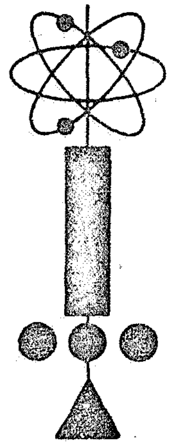
CONTRACT NO. NAS3-2533

NASA DOCUMENT NO. CR-54349
GE DOCUMENT NO. 65SD4296 (VOL 1)
GE DOCUMENT NO. 65SD4297 (VOL 2)
GE DOCUMENT NO. 65SD4298 (VOL 3)



NAVIGATOR
STUDY OF ELECTRIC PROPULSION FOR UNMANNED SCIENTIFIC MISSIONS
VOLUME 3
SPACECRAFT PERFORMANCE AND SUMMARY

Prepared For
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
LEWIS RESEARCH CENTER
21000 BROOKPARK ROAD
CLEVELAND, OHIO

15 JULY 1965

GENERAL ELECTRIC
MISSILE AND SPACE DIVISION

CONTRACT NO. NAS3-2533

NASA DOCUMENT NO. CR-54349
GE DOCUMENT NO. 65SD4296 (VOL 1)
GE DOCUMENT NO. 65SD4297 (VOL 2)
GE DOCUMENT NO. 65SD4298 (VOL 3)

NAVIGATOR
STUDY OF ELECTRIC PROPULSION FOR UNMANNED SCIENTIFIC MISSIONS
VOLUME 3
SPACECRAFT PERFORMANCE AND SUMMARY

Prepared For
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
LEWIS RESEARCH CENTER
21000 BROOKPARK ROAD
CLEVELAND, OHIO

Attn: NUCLEAR POWER TECHNOLOGY BRANCH
ROBERT J. DENINGTON
TECHNICAL MANAGEMENT

Approved by: H Brown
H. Brown, Manager
Mission Analysis and Evaluation

Approved by:
B. J. Tharpe, Program Manager

Approved by:
T. F. Widmer, Manager
Advanced Nuclear Programs

Approved by:
E. Ray, Manager
Advanced Nuclear Systems Engineering

GENERAL ELECTRIC
MISSILE AND SPACE DIVISION

TABLE OF CONTENTS

TABLE OF CONTENTS (Cont'd)

LIST OF ILLUSTRATIONS

LIST OF TABLES

1. INTRODUCTION

This Topical Report presents the results of studies performed by the General Electric Missile and Space Division during the nine month extension of Contract NAS 3-2533, Study of Electric Propulsion for Unmanned Scientific Missions. Five reports* were issued in the original contract under the title of Research on Spacecraft and Powerplant Integration Problems.

This program, "NAVIGATOR," was initiated by GE-MSD under contract to the NASA Lewis Research Center. The program objective is to determine requirements for the nuclear-electric power generating systems required in the NASA unmanned scientific probe missions throughout the solar system, which are beyond the capabilities of the presently envisioned chemical rocket propelled vehicles.

In the original contract, consideration was given to vehicles powered by advanced nuclear powerplants and thrusters that began electric propulsion from earth orbit or at escape velocity. In the contract extension, consideration is given to earlier powerplants with modest technology requirements that are launched to escape and beyond to reduce the trip time. Thus, the two studies combine to span a large spectrum of nuclear electric propelled vehicle capabilities.

The results obtained in the current nine month study extension are presented in three volumes. These are:

---

* 1. 63SD760, First Quarterly Report, 26 April to 26 July, 1963;
2. 63SD886, Second Quarterly Report, 26 July to 26 October, 1963;
3. 64SD505, Mission Analysis Topical Report, February 26, 1964;
4. 64SD700, Third and Fourth Quarterly Report, 26 October 1963 to 26 April, 1964; and
5. 64SD892, Spacecraft Analysis Topical Report, July 24, 1964, NASA Document CR-54159.

- Volume 1 - Volume 1 (CR-54324) encompasses the mission analyses. It describes the analytical techniques applied in the analyses; it presents the vehicle and powerplant requirements in terms of trip time, power level, and payload for optimum orbiter and flyby missions as accomplished by electrically propelled spacecraft; and it presents the payload and trip time capabilities for chemical and chemical plus nuclearly propelled spacecraft for the same missions.
- Volume 2 - Volume 2 (CR-54348, Classified CRD) compares first generation nuclear powerplants based upon an uprated SNAP-8 Mercury/Rankine Cycle, the Brayton Cycle, and the Potassium/Rankine Cycle power systems. The comparison shows that only the Potassium/Rankine system can result in a powerplant of sufficiently low weight to competitively accomplish a useful scientific mission. Payloads for the vehicles and operating modes for the powerplants are discussed.
- Volume 3 - the present Volume (CR-54349) relates the mission requirements described in Volume 1 to the power system/vehicle capabilities discussed in Volume 2. It thus defines those missions that can be accomplished with powerplants of both early and forseeable technology and it compares the capabilities of nuclear electric propelled spacecraft with those of chemically propelled spacecraft and with those of chemical plus nuclear rocket propelled spacecraft.

The results show that there are useful scientific missions that can be accomplished more advantageously with nuclear electric vehicles of even modest specific weights than with vehicles utilizing either all chemical or chemical plus nuclear rocket propulsion. A process of orderly development is, therefore, available whereby the early powerplants can be used for the near planet missions and the experience gained in these applications used to decrease powerplant specific weights. These improvements will provide power of less than 30 lb/KWe as required for more difficult planetary exploration.

2. SUMMARY

This section summarizes the results presented in Volume 1 and 2 in addition to the summary comparison of spacecraft performance described in this volume.

2.1 VOLUME 1: MISSION ANALYSIS

The Navigator mission studies previously reported were concerned with the capabilities of advanced nuclear powerplant and thruster technology. These studies involved the investigation of planetary orbiter missions to each of the planets of the solar system except Mars and Venus, a solar probe, and an out-of-the-ecliptic mission. Payload requirements for providing planetary and satellite soft landing capsules, high resolution radar, television, and a number of sophisticated scientific experiments were identified and assumed for each of the NAVIGATOR missions. The studies were limited to the use of a single chemical propulsion stage beyond orbit and, in general, used a propulsion-coast-propulsion profile for the nuclear-electric phase of each mission. Although the results illustrated the suitability of a 1 MW powerplant for most of the NAVIGATOR missions investigated, propulsion requirements ranging from 3000 to 25,000 hours were obtained with coasting requirements up to 20,000 hours. Only three of the missions investigated could be performed within 10,000 hours of propulsion.

The present study considers an "Early-technology" powerplant involving powerplant specific weights up to 70 lbs/kwe, power levels of 100 to 400 kwe, and operating lifetimes up to 15,000 hours. Planetary fly-by missions are considered in addition to the previous orbiter missions. The number of initial chemical propulsion stages is increased to two stages to provide a maximum high thrust characteristic velocity of 40,000 fps as a means for reducing both propulsion time and trip time requirements. The mission profile is altered to include only a single continuous electrical propulsion period as a means of eliminating the long intermediate coast period between the two periods of operation at full power. The scope of the study is expanded to consider the effects of variable specific impulse operation and to obtain chemical and nuclear propulsion mission capabilities with which to compare the above nuclear-electric propulsion results.

2.1.1 SPACECRAFT PERFORMANCE

The present study develops a set of generalized performance characteristics which can be used to obtain low thrust propulsion requirements for the heliocentric phase of optimum *fly-by and orbiter missions in the solar system. These data are used as the basis for generating a series of mission performance maps for each of the NAVIGATOR missions. These maps show the variation in mission payload capabilities for each mission as a function of total trip time and powerplant specific weight. Auxiliary parameters displayed on these maps include propulsion time, power rating, specific impulse, and rocket characteristic velocity. Comparable data is presented, for each mission, illustrating the performance capabilities of chemical and nuclear propulsion for the NAVIGATOR type missions.

2.1.2 FLYBY MISSIONS

Fly-by performance data is given for the solar probe, Mercury, Asteroid, Jupiter, and Saturn missions for the Saturn 1B booster and an electric propulsion stage. These data are based upon the use of the SIB to earth orbit and nuclear-electric propulsion from earth orbit. The solar probe and Mercury fly-by data assumes a minimum ion engine specific impulse of 3,000 seconds and cover a propulsion time range of 1,000 to 5,000 hours. Attractive payloads are obtained for the Asteroid probe and the Jupiter fly-by for the complete range of powerplant specific weights with less than 15,000 hours propulsion time. The Saturn fly-by, on the other hand, requires in excess of 20,000 hours propulsion time with powerplant specific weights of 50 lb/kw or greater. It represents, therefore the limiting case for application of the Saturn 1B to the NAVIGATOR missions.

Performance data is repeated for the Saturn, Uranus, Neptune, and Pluto fly-bys and Out-of-the-Ecliptic-Probe with the Saturn 5 booster and an additional one to two stages of high thrust propulsion. These data are shown for operation at 10,000 and 15,000 hours propulsion time. The trip time requirements for these missions range from 10,000 to 38,000 hours and the optimum specific impulse from 4,000 to 7,500 seconds.

---

* Optimum is defined as maximum payload at a given trip-time.

2.1.3 ORBITER MISSIONS

Orbiter performance data is shown for the Saturn V booster with one or two stages of high thrust propulsion. A minimum specific impulse of 3000 seconds is used for the Mercury, Venus, and Mars orbiters. Propulsion time requirements for these missions range from 1,000 to 5,000 hours. The optimum specific impulse ranges from 3,000 to 16,000 seconds for the remaining orbiter missions. The corresponding propulsion times are 4,000 to 30,000 hours.

2.1.4 VARIABLE SPECIFIC IMPULSE

Investigations of the effects of variable specific impulse showed a 10% performance improvement for impulse variations of 10 to 15% for the relatively easy fly-by and orbiter missions. This improvement disappears for the more difficult missions.

2.2 VOLUME 2: COMPARISON OF NUCLEAR POWER SYSTEMS

In Volume 2 (Classified CRD), parametric data are provided for the major components of the three nuclear power systems listed below:

- An up-rated SNAP-8 Power System,
- A two-loop Brayton Cycle System with a NaK cooled, Uranium carbide fueled reactor, and
- A three-loop Potassium/Rankine Cycle System with the same reactor.

These data cover the range of power levels from 40 to 400 KWe and a wide range of power system operational conditions. Each major subsystem or component is examined separately and the weights determined are used to provide reliable estimates of total powerplant weight. At equivalent technologies, the weights for the three system are compared, including consideration of operational factors. The system weights and operating conditions are based on a conservative level of "first-generation" technology for nuclear systems. In particular, temperatures are limited to those that are compatible with non-refractory metals, moderate component efficiencies are assumed, presently fabricable radiator

... are used, NaK is the reactor coolant, and reactor temperature is limited to ... those of SNAP-50. These conservative conditions yield powerplant specific ... higher than the 20 to 30 lb/KWe usually determined; however, the weights are likely ... for the "first-generation" powerplants.

... UC FUELED REACTOR

The NaK cooled, uranium carbide fueled reactor provides a maximum coolant outlet temperature of 1850^{o}F and an energy output of 14,000 MW-hr with the minimum size reactor. The reactor is burnup limited and the core size must be increased for greater outputs.

... SHIELD WEIGHT

Shield weights were determined for a range of reactor sizes, power levels, operating ..., shielding angles, payload dose and payload separation distances. Generally, in ... scientific spacecraft, the shield is 10 to 20% of the total powerplant weight.

... POWER CONDITIONING

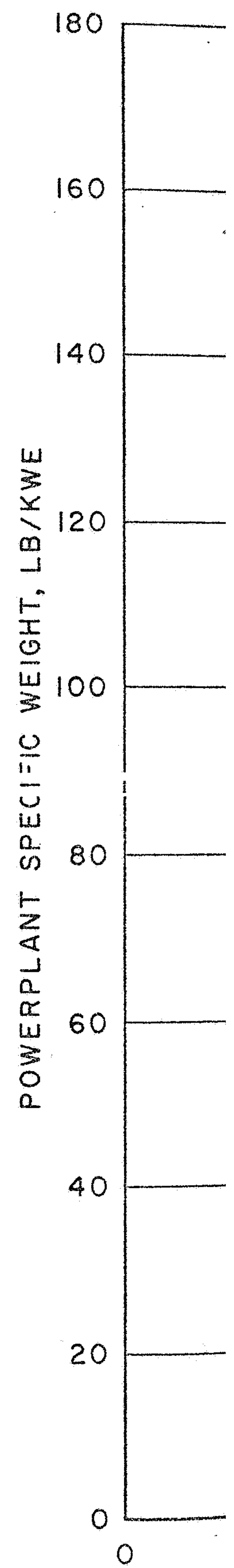
Power conditioning is evaluated for the MHD Arc jet, the contact ionization, and the Elec... electric propulsion engines. These engines can provide specific im... the required range of 2500 to 7000 seconds. The specific weight of the power ... equipment ranges from 1 to 3 lb/KWe in inverse relationship to the specific ...

... POWER SYSTEM COMPARISON

... auxiliary cooling, reactor, power conversion system and primary and ... were examined for all three power systems. Based on an examination ... temperature limitations and the state of development, the weight of ... powerplant can be defined with the result shown in Figure 2.2-1.

	SNAP-8 SYSTEM	BRAYTON CYCLE SYSTEM	PO RANK
REACTOR TYPE	SNAP-8	UC FUELED, NAK COOLED	UC F NAK
MAXIMUM REACTOR COOLANT TEMPERATURE	1300°F	1850°F	18
TURBINE INLET TEMPERATURE	1250°F	1800°F	17
TURBINE EFFICIENCY	60 %	85%	
COMPRESSOR EFFICIENCY	-----	80%	
RADIATOR RELIABILITY (15,000 HOURS)	0.9	0.9	
SINK TEMPERATURE	26°F	26°F	
PRIMARY RADIATOR MATERIAL	Al/STAINLESS STEEL	Al/STAINLESS STEEL	COPF
POWER CONVERSION EQUIPMENT	PARTIALLY REDUNDANT	NON REDUNDANT	RE
PAYLOAD SEPARATION DISTANCE	100 FT.	100 FT.	
SHIELD HALF-ANGLE	30°	30°	
PAYLOAD DOSE			
NEUTRON	10^{11} NVT	10^{11} NVT	
GAMMA	10^{6} RAD	10^{6} RAD	
FULL POWER OPERATION LIFETIME	10,000 HOURS	10,000 HOURS	1C

	SNAP-8 SYSTEM	BRAYTON CYCLE SYSTEM	POTASSIUM/ RANKINE SYSTEM
₹ TYPE	SNAP-8	UC FUELED, NAK COOLED	UC FUELED, NAK COOLED
M REACTOR T TEMPERATURE	1300°F	1850°F	1850°F
E INLET IATURE	1250°F	1800°F	1700°F
E FICIENCY	60 %	85%	70%
:SSOR EFFICIENCY	-----	80%	-----
R RELIABILITY OO HOURS)	0.9	0.9	0.9
EMPERATURE	26°F	26°F	26°F
Y RADIATOR ERIAL	Al/STAINLESS STEEL	Al/STAINLESS STEEL	COPPER/STAINLESS STEEL
CONVERSION PMENT	PARTIALLY REDUNDANT	NON-REDUNDANT	NON-REDUNDANT
) SEPARATION ANCE	100 FT.	100 FT.	80 FT.
HALF-ANGLE	30°	30°	15°
D DOSE			
'RON	10^{11} NVT	10^{11} NVT	10^{11} NVT
MMA	10^{6} RAD	10^{6} RAD	10^{6} RAD
OWER OPERATION FETIME	10,000 HOURS	10,000 HOURS	10,000 HOURS



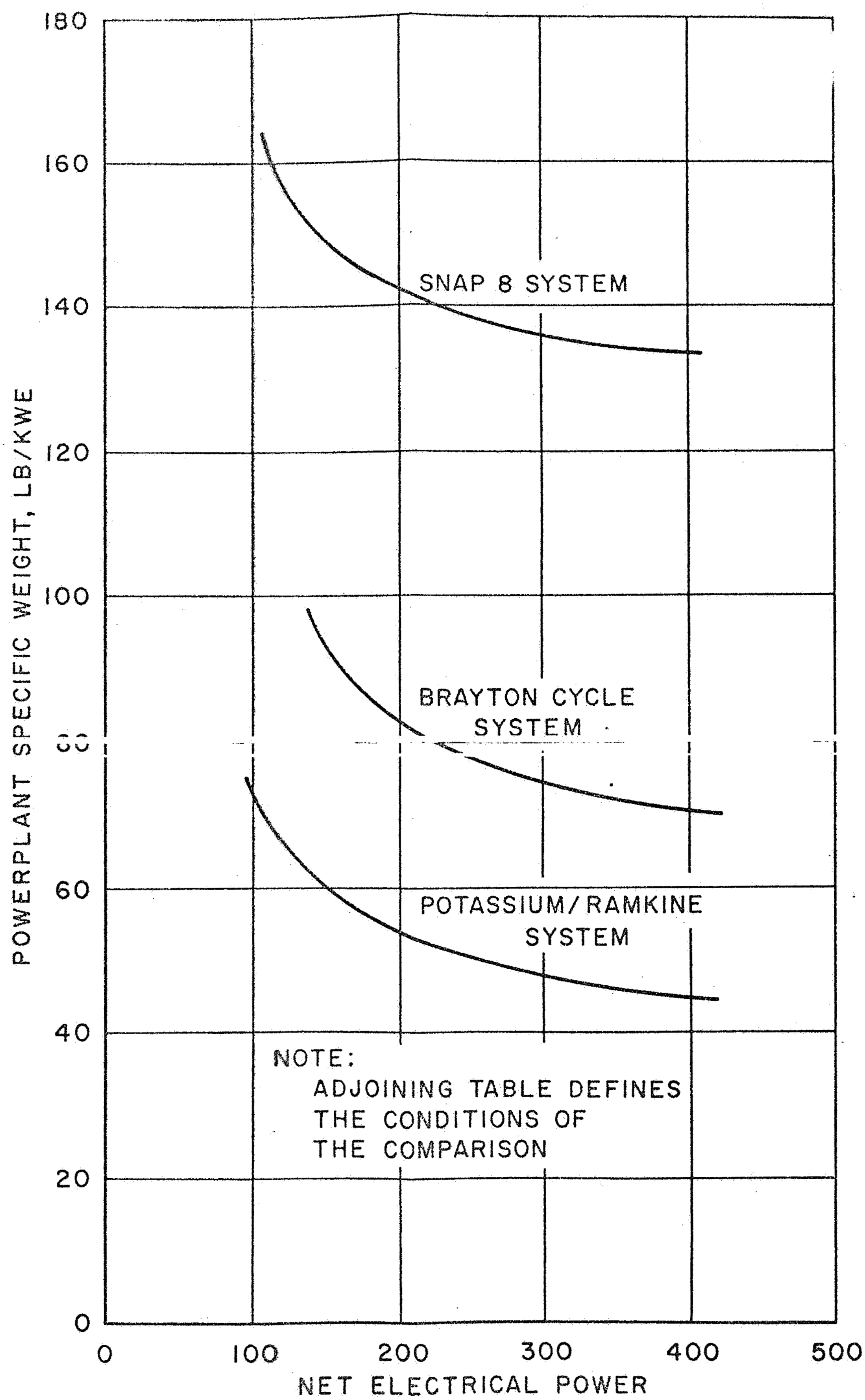


Figure 2.2-1. Comparison of System Specific Weights

In the comparisons, the Potassium/Rankine system is evaluated to be equivalent or superior in all respects except that the power conversion equipment for the Brayton Cycle system is likely to be less difficult to develop and the gas radiators are not susceptible to freezing. On this latter point, however, it may be necessary to substitute a segmented and redundant liquid metal radiator for the gas radiator in order to obtain high reliability. Therefore, this apparent advantage may not, in fact, be present. The advantages of the Potassium/Rankine system (smaller radiators that result in ease of integration, fixed radiator construction, and lower specific weight) more than balance the advantages of the gas power conversion equipment and, consequently, the Potassium/Rankine power systems is evaluated to be nearest optimum for the NAVIGATOR missions.

2.2.5 POWER SYSTEM DESIGN

For the Potassium/Rankine Power System, consideration is given to the launch and initial start-up, to methods of shutdown and restart, and to methods of providing electrical power during the coast periods of the NAVIGATOR missions. The problem of maintaining the liquid metal systems, especially the radiators, in a liquid state prior to start-up and during the coast period is particularly important. This examination indicates that the following features will be required in the powerplants:

- The radiators will be thermally shrouded at launch and for the first start-up. However, in further power reductions, radiator temperatures will be maintained above freezing by rejection of reactor heat.

  A chemical auxiliary power unit sufficient for start-up and a 10 hour "wait" period prior to start-up will be required.

- During the coast periods of both the orbiter and flyby missions, the reactor will be operated at low power and reactor energy will be transferred to the radiators via an auxiliary heat exchanger loop to prevent radiator freeze-up.

- The auxiliary loop will include thermoelectric converters to produce several kilowatts of power. This is sufficient for "house-keeping" power and for minimum communications with earth.

- The power system for flyby missions will include re-start capability for either a "dry" or a "wet" start-up.

- The operational sequence for the flyby missions is:

  - Launch and "hold" for 10 hours,

  - Start-up and operation at full power for propulsion,

  - Shutdown of the dynamic system with continued reactor operation at low power and with electrical power generation via the thermoelectric converters during coast, and,

  - Restart of the dynamic power conversion system for payload power.

- The operational sequence for the orbiter missions is:

  - Launch and "hold" for 10 hours,

  - Start-up and operation at low power with electrical power generation via the thermoelectric converters during coast, and

  - Full power operation with the dynamic system for propulsion followed by payload power.

Example designs for Potassium/Rankine Powerplants were prepared for power levels of 160, 240 and 320 KWe to confirm the validity of the parametric evaluations and to illustrate the characteristics of powerplants over the power range of interest. The powerplants are designed to mate with the 260 inch diameter S-IV-B stage of the Saturn V vehicle. The powerplants package within the available payload envelope with ease, allowing more than adequate space for the payload.

2.2.6 SCIENTIFIC PAYLOADS

The NAVIGATOR payloads include:

- Landing Capsules
  - Surface landing capsules for the small planets and the large satellites
  - Atmospheric probes for the major planets.
- Orbiter Scientific Instrumentation
  - Field and Particle Detectors
  - TV and Optics
  - Radar
- "In-transit" Scientific Instrumentation
  - Field and Particle detectors
  - Solar and cosmic spectra
- Communications and Data Handling Equipment

There an extremely large number of scientific measurements that can desirably be made in the planetary and solar environments with instruments of modest weight (a few pounds) and with modest power requirements (1 to 50 watts per experiment).

Consequently, instrument packages of a few hundred pounds and requiring a few hundred watts of power can accomplish a significant mission. For maximum benefit, however, this package must be delivered to a planetary or satellite surface which will require 2000 to 4000 pounds of additional weight, and the data must be communicated to Earth which will require 10's to 100's of kilowatts of power.

Additionally, experiments that involve visual images (e.g. television or radar pictures) of the outer planets require payloads of thousands of pounds and power levels of tens of kilowatts. With adequate power, any total payload capability can be utilized to provide increased reliability through redundancy of experiments, to provide additional experimental sophistication, and to provide additional communications capability. This latter capability is particularly important for the outer planet missions in which the Navigator will likely find application.

Designs for typical payloads covering the range of 3, 000 to 12, 000 pounds were prepared and various methods of deploying the payload from the power system were considered. The payloads are designed to interface with any of the three Potassium/Rankine Powerplants, thus providing interchangeability between payloads and powerplants. Typical combinations of Powerplant and Payload for the NAVIGATOR missions are shown in Figures 2.2-2 and 2.2-3. A 160 KWe power is shown with a 3, 000 pound payload in Figure 2.2-2 and a 240 KWe system is shown with a 12, 000 pound payload in Figure 2.2-3. It is possible to interchange the power systems and the payloads to meet the power/payload requirements of the various mission.

2.3 VOLUME 3: SPACECRAFT COMPARISON

As shown by the summaries given above, Volume 1 describes the payload capabilities for the all-chemical and chemical-plus-nuclear rocket propulsion systems. It also presents the payloads for electric propulsion vehicles with powerplants of various specific weights.

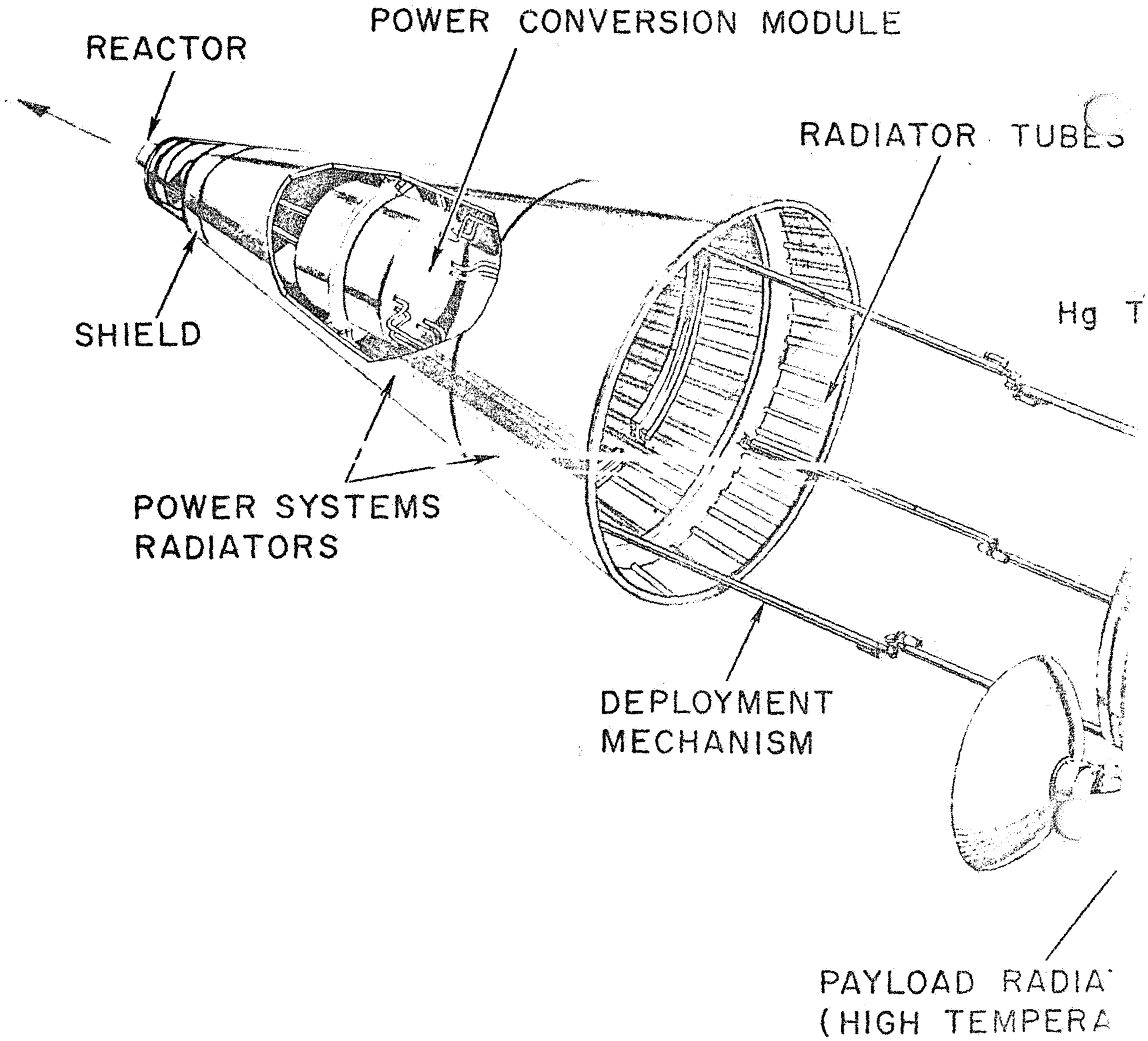
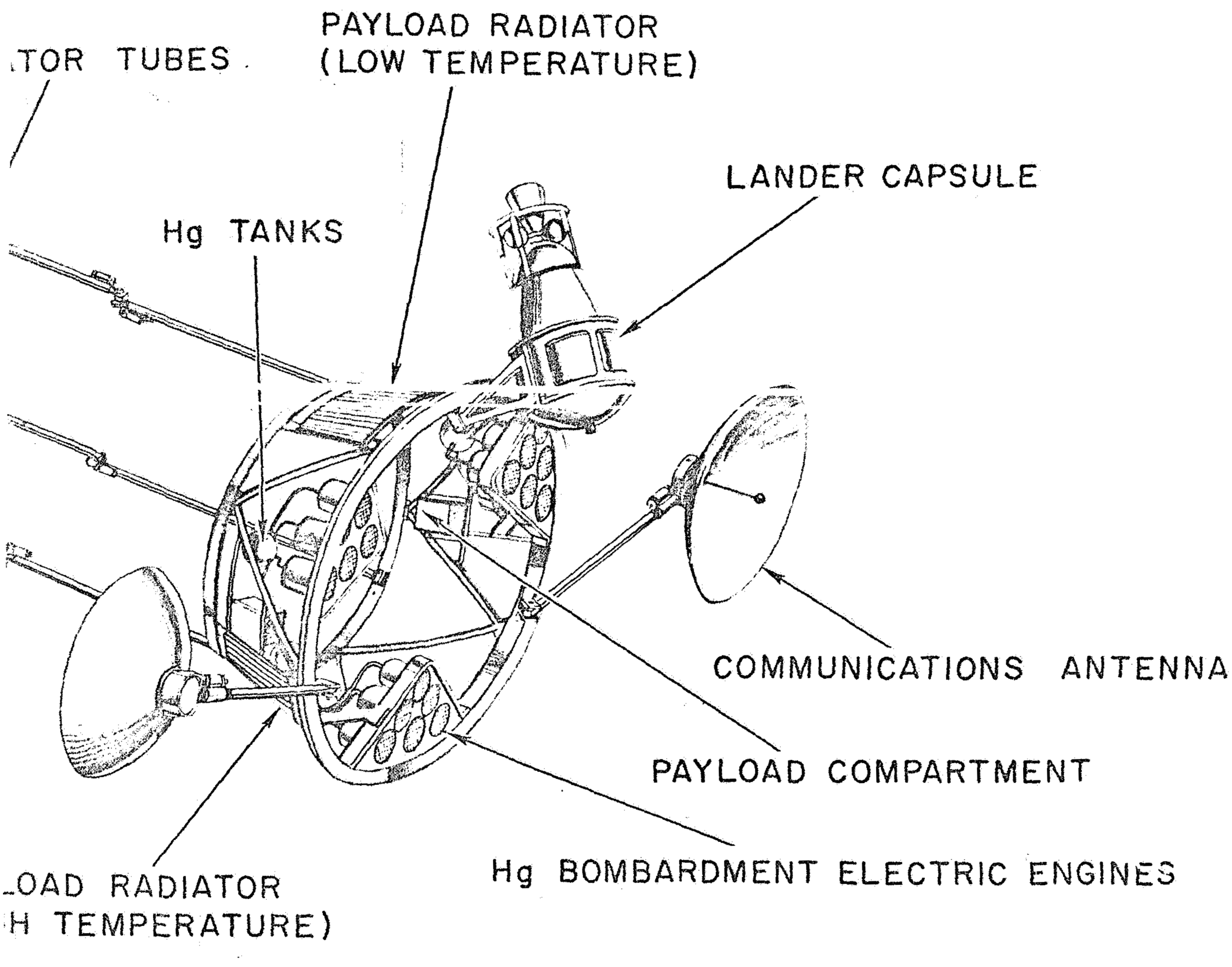
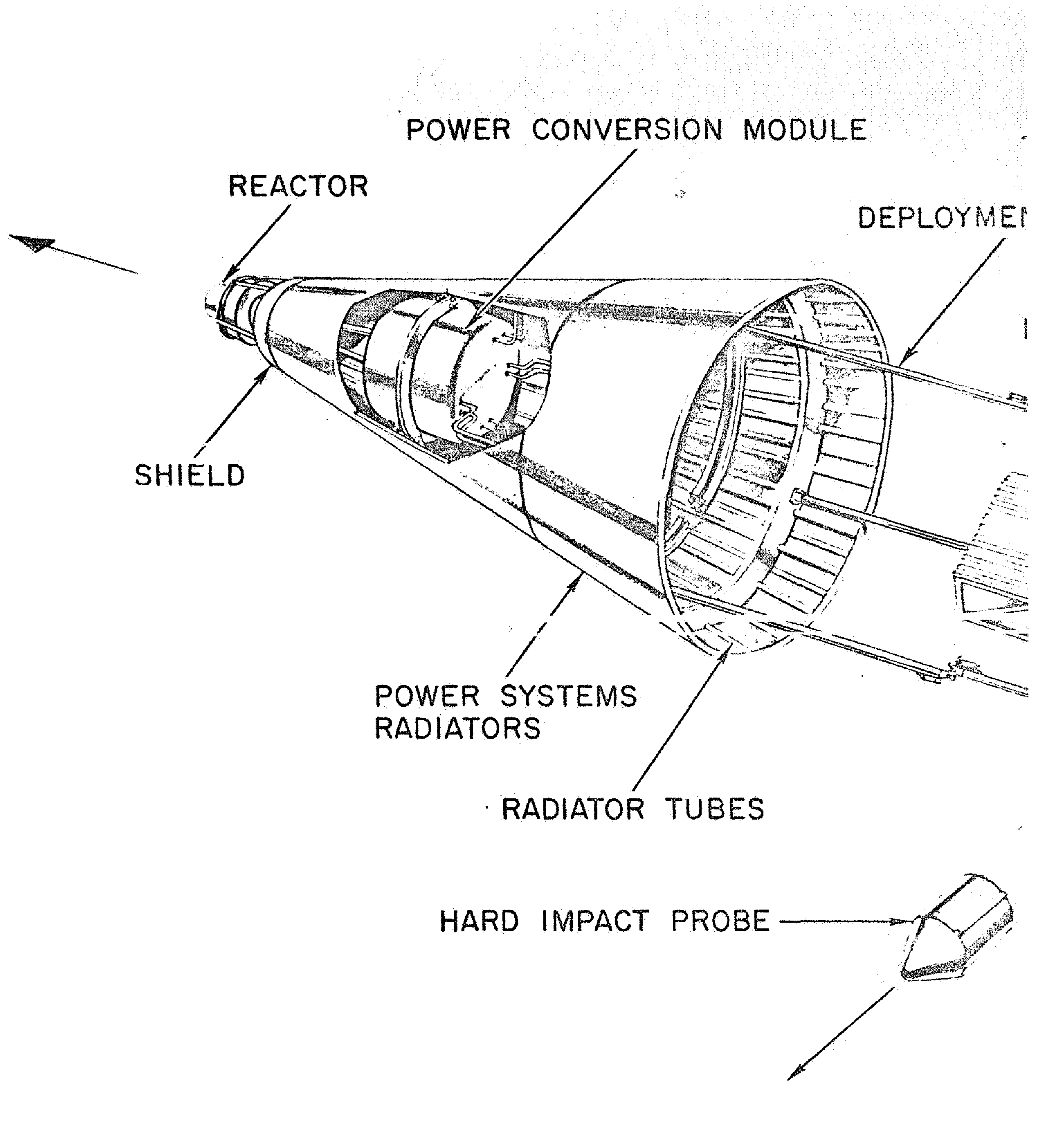


Figure 2.2-2. Navigator Vehicle, 160 KWe Power System, 3000 Pound Payload



ound Payload


POWER CONVERSION MODULE
REACTOR
DEPLOYMEN
SHIELD
POWER SYSTEMS
RADIATORS
RADIATOR TUBES
HARD IMPACT PROBE

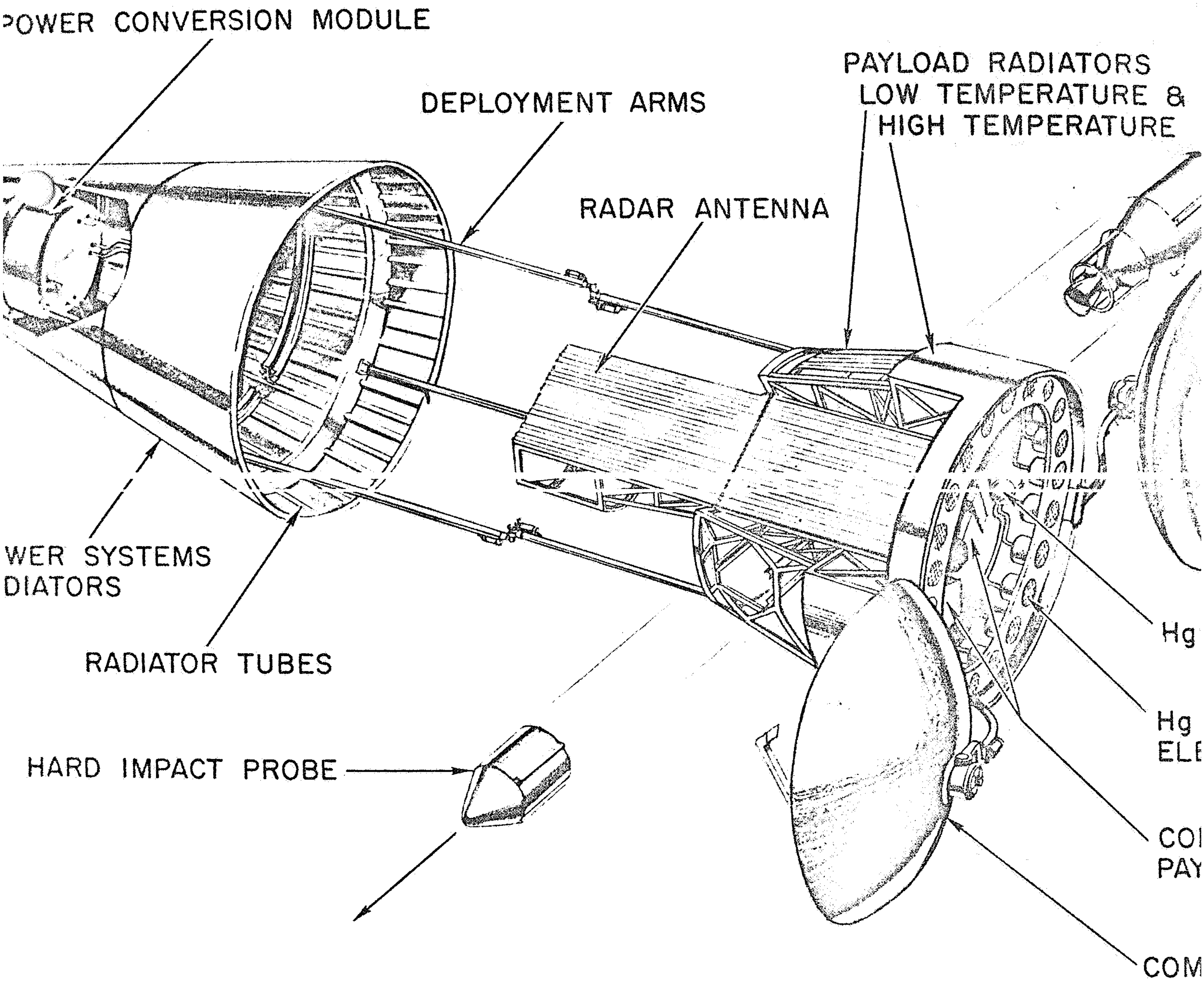


Figure 2.2-3. Navigator Vehicle, 240

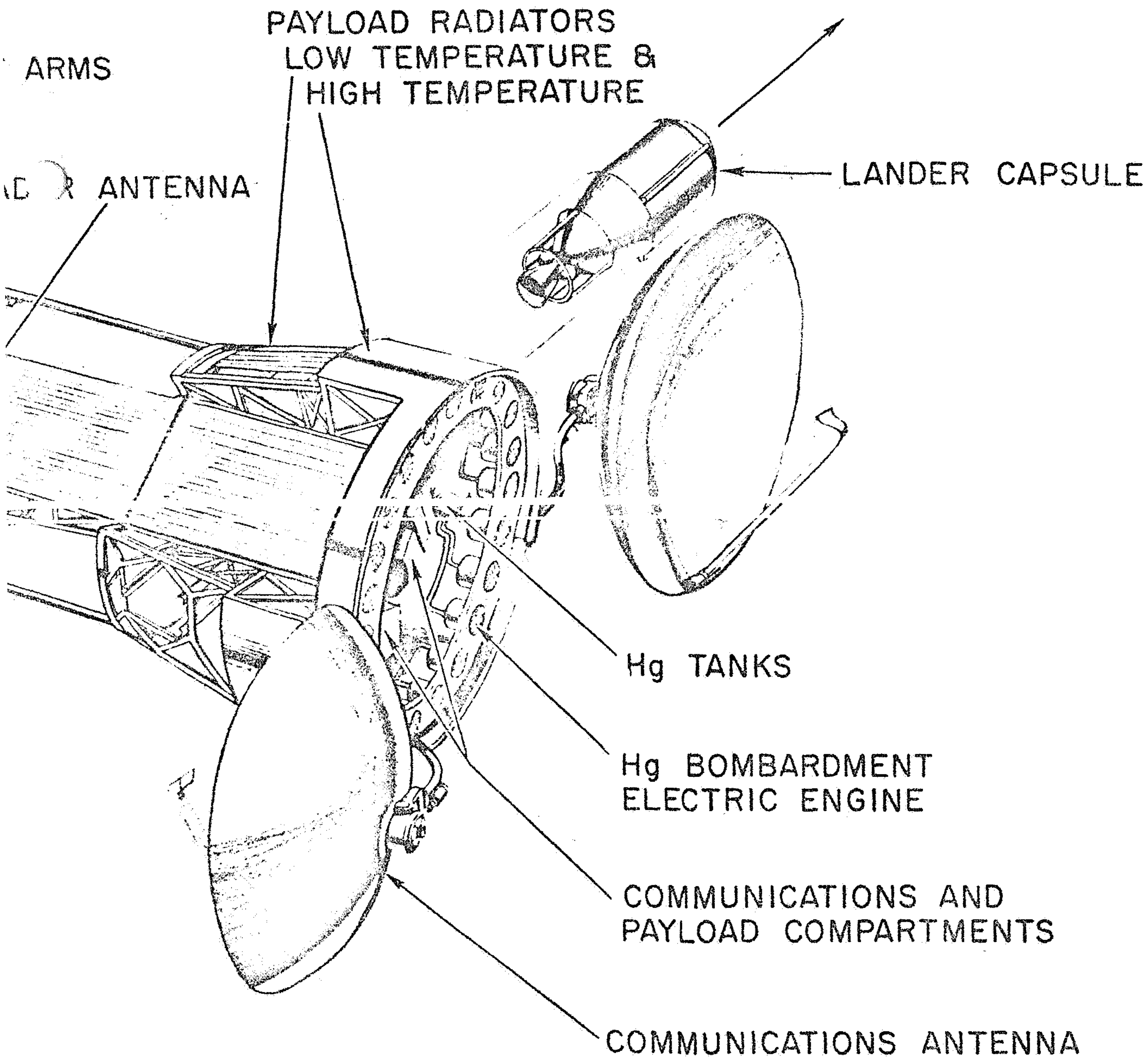


Figure 2.2-3. Navigator Vehicle, 240 KWe Power System, 12,000 Pound Payload

Volume 3 also examines several potential methods by which the performance of electric propulsion vehicles may be improved. These include the:

- Use of two optimum electric propulsion periods after the vehicle is launched beyond escape by the multistage S-V, and

- Use of a nuclear rocket stage in conjunction with the electric propulsion vehicle.

Both of the mission modifications result in significant improvements in trip time, operating time, and/or payload.

3. VEHICLE PROPULSION SYSTEMS

3.1 RELATION OF NON-ELECTRIC AND ELECTRIC PROPULSION VEHICLE PAYLOADS

In the comparison of payload capabilities, it is necessary to relate the gross payload* of the chemical or chemical plus nuclear rocket vehicles to equivalent payloads for electric propulsion vehicles. This is necessary because in the electric propulsion vehicles (EPV), the propulsion power system can be used for payload power.

It is also necessary to determine the minimum gross payload that is required to deliver the minimum acceptable scientific payload.

3.1.1 PAYLOADS FOR "FLYBY" MISSIONS

To adjust Non-EPV gross payloads for the inclusion of a power system, a minimum scientific payload and communications requirement is defined. Volume 2, Section 5, shows that a scientific instrument package of at least 480 pounds will be required for the "in-transit" and the planetary scientific sensors. The data output of these instruments can vary widely depending upon the particular experiments. If either radar or TV systems are included, the information rate will be 10^6 to 10^7 bits/sec; whereas for other types of experiments, the information rate may be as low as 2×10^3 bits/sec**. The higher rate will favor the EPV vehicles as many kilowatts of power will be required; however, to assure a conservative comparison, the lower data rate is assumed.

---

*Gross payload is the total delivered weight exclusive of the propulsion system. The net payload is that available for scientific experiments after allowance for structure, guidance and control, stabilization, communications, power and other necessary subsystems.

**"Voyager Spacecraft System Study", General Electric Company, Missile and Space Division, August 1964, Document No. 64SD933.

Assuming that the Deep Space Information Facility will be available only 2 hours per day, a minimum communications rate of 24,000 bits/sec will be required for a single transmittal of the data generated by the scientific instruments. Figure 5.3-1 (Volume 2, CR-54348) indicates that a power input of 10 KWe will be required for this communications rate from Saturn with a 20 foot diameter sending antenna. Figure 3.1-1 indicates that a 10 KWe power system will weigh about 4600 pounds and the relationship shown in Table 3.1-1 is obtained between EPV and Non-EPV for Saturn.

Table 3.1-1 Flyby Payload for Saturn

	Electric Propulsion Vehicle	Non-Electric Propulsion Vehicle
Scientific Instruments, lbs.	480	480
Power System, lbs.	——	4600
Equivalent Payload Weights	480 lbs.	5080 lbs.

The 480 pound and 5080 pound payloads provide equivalent scientific information; however, neither of these payloads is sufficient for the mission. Additional payload is required for at least the 20 foot diameter communications antennas (350 pounds), for the communications transmitters (15 lb/KWe), for high temperature payload cooling (8 lb/KWt), for guidance and stabilization systems (500 lbs.), and for structure (10% of gross payload). This additional equipment results in the minimum total payload weights shown in Table 3.1-2. The payloads of 1710 pounds and 6820 pounds are the minimum necessary to support the 480 pounds of scientific instruments delivered by EPV and Non-EPV, respectively. The results for the other planetary Flyby missions are given in Table 3.1-2 also.

The disparity between payloads for EPV and Non-EPV reduces with increased payload size as shown by Figure 3.1-2. This figure relates gross Non-EPV payload to equivalent EPV payload. The beginning point for each planet is the minimum payload given in Table 3.1-2. The decreased disparity of payloads is a result of the proportionate increase in payload and communications weight for both types of vehicles, the relatively small increase in power system weight for the Non-EPV vehicles and the overshadowing of the guidance and stabilization system weight.

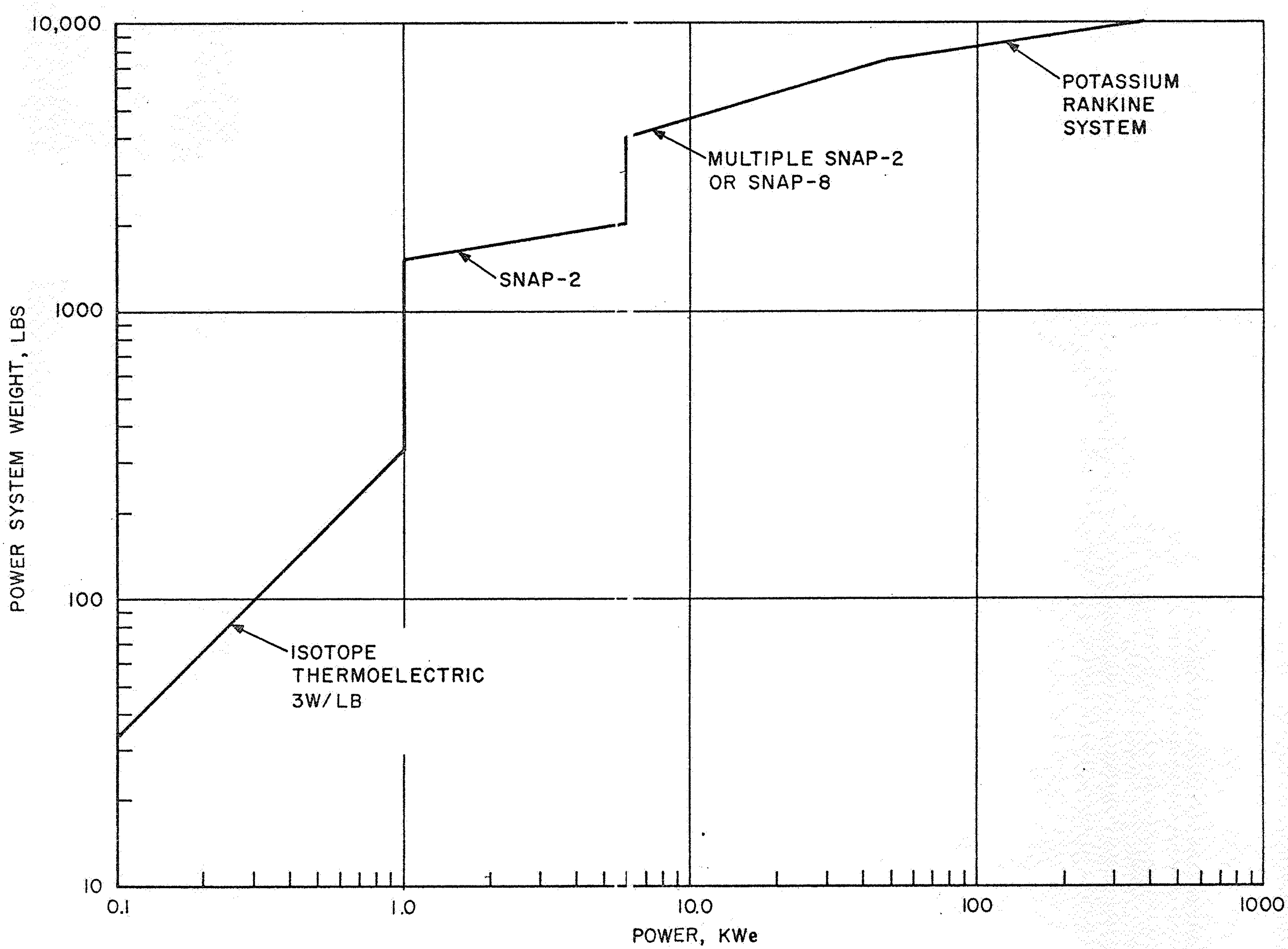


Figure 3.1-1. Power System Weight

Table 3.1-2 Minin

Planet	Communications Power, KWe	Scientific Instruments	Guidance & Stabilization	Communication Antenna	Cc
Mercury, Solar Probe, Out-of-the-ecliptic	0.1	480	500	350	
Asteroids	1.0	480	500	350	
Jupiter	3	480	500	350	
Saturn	10	480	500	350	
Uranus	30	480	500	350	
Neptune & Pluto	100	480	500	350	

.1-2 Minimum Payloads for Flyby Vehicles
Weight, Lbs.

ion	Communication Transmitter	High Temp. Cooling	Power System	Structure	Non-Electric Propulsion Vehicle Payload Weight	Equivalent Electric Propulsion Vehicle Payload Weight
	30	10	30	150	1,550	1,520
	30	10	300	190	1,860	1,520
	45	18	1,500	320	3,210	1.540
	150	60	4,000	680	6,820	1,710
	450	180	6,500	940	9,400	2,180
	1,500	600	8,200	1,290	12,920 Lbs.	3,770 lbs.

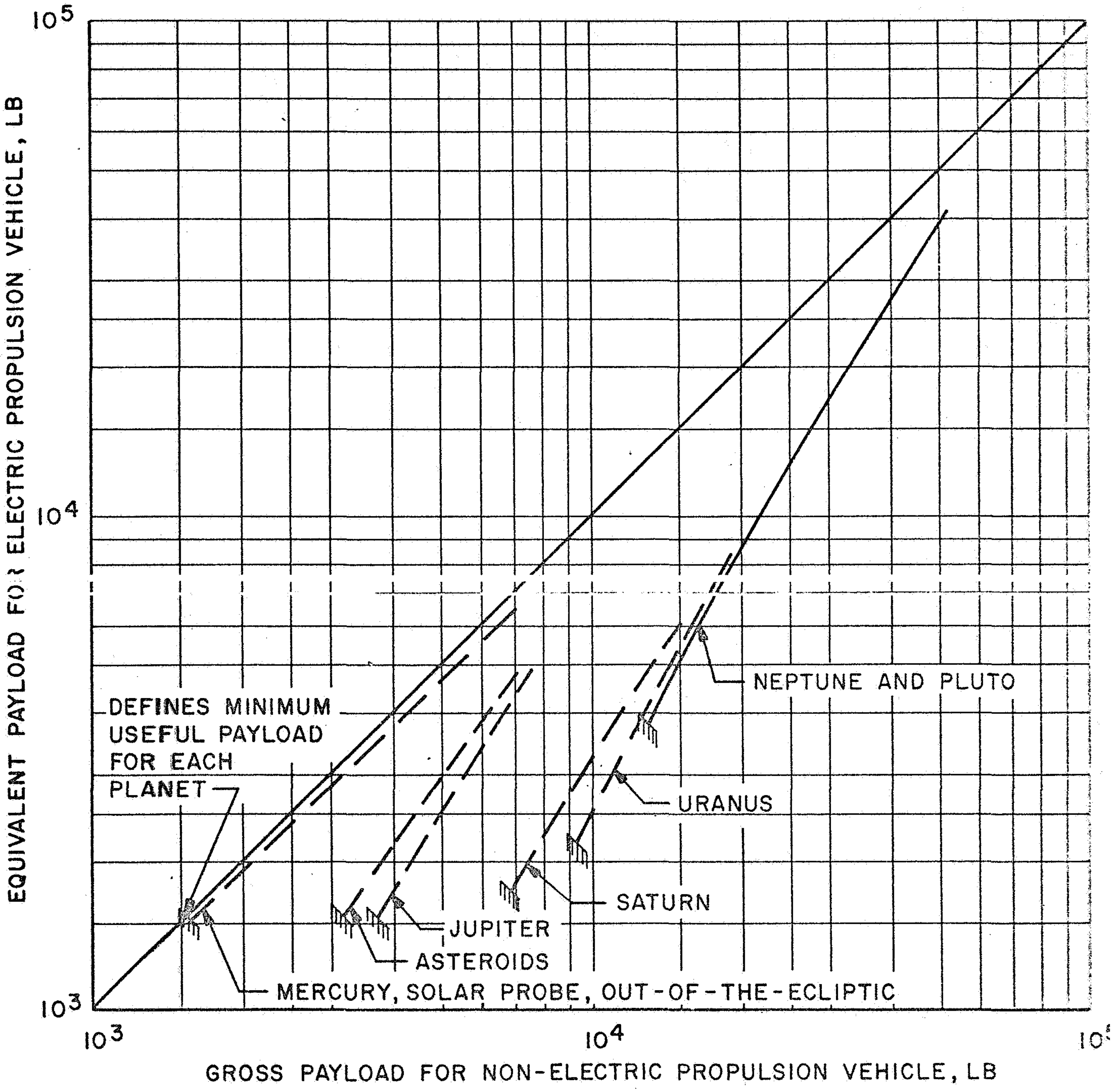


Figure 3.1-2. Relation of Gross and Equivalent Payloads for Flyby Missions

The small disparity for the mercury, solar probe, and out-of-the-eclyptic missions could be significantly increased if radar is included in the payload or if a higher communication rate is necessary.

The minimum payload weights given in Table 3.1-2 can be combined with the gross payload capabilities of Non-EPV to give the result shown in Figure 3.1-3 and 3.1-4. In Figure 3.1-3, the minimum payloads define an operational area of exclusion on the gross payload in which the all chemical, high thrust propulsion system cannot deliver the minimum payload package for the 5 x 10^6 mile solar probe. However, the payload is more than adequate for the other probes and minor planet flybys. Figure 3.1-4 shows the similar relationship for the major planet Flyby missions. The Saturn IB is adequate for the Jupiter mission although the trip time can be reduced significantly with the Saturn V all chemical vehicle. The Saturn and Uranus missions require the Saturn V all chemical system and the Pluto and Uranus missions require the Saturn V vehicle with a nuclear rocket stage to obtain near reasonable trip times with adequate payloads.

Figure 3.1-2 can be used to convert the gross Non-EPV payload weights given in Volume 1 to equivalent EPV payloads with the result shown in Figures 3.1-5 and 3.1-6. These payloads can be compared directly with those given for the electric propulsion vehicles.

3.1.2 PAYLOADS FOR "ORBITER" MISSIONS

The technique used to relate Non-EPV and EPV payloads and to determine minimum payloads for the "Orbiters" is similar to that used for the Flyby missions. However, since the "Orbiters" will be in close proximity to the target planet for a much greater time and are expected to provide more useful information on the planets and their satellites, the minimum scientific payload is increased from 480 pounds to 3655 pounds. This allows the inclusion of a soft-impact landing capsule (2600 pounds)* and a hard-impact atmospheric probe (575 pounds).

*"Research on Spacecraft and Powerplant Integration Problems", General Electric Company, Missile and Space Division, Feb. 1964, Document No. 64SD505.

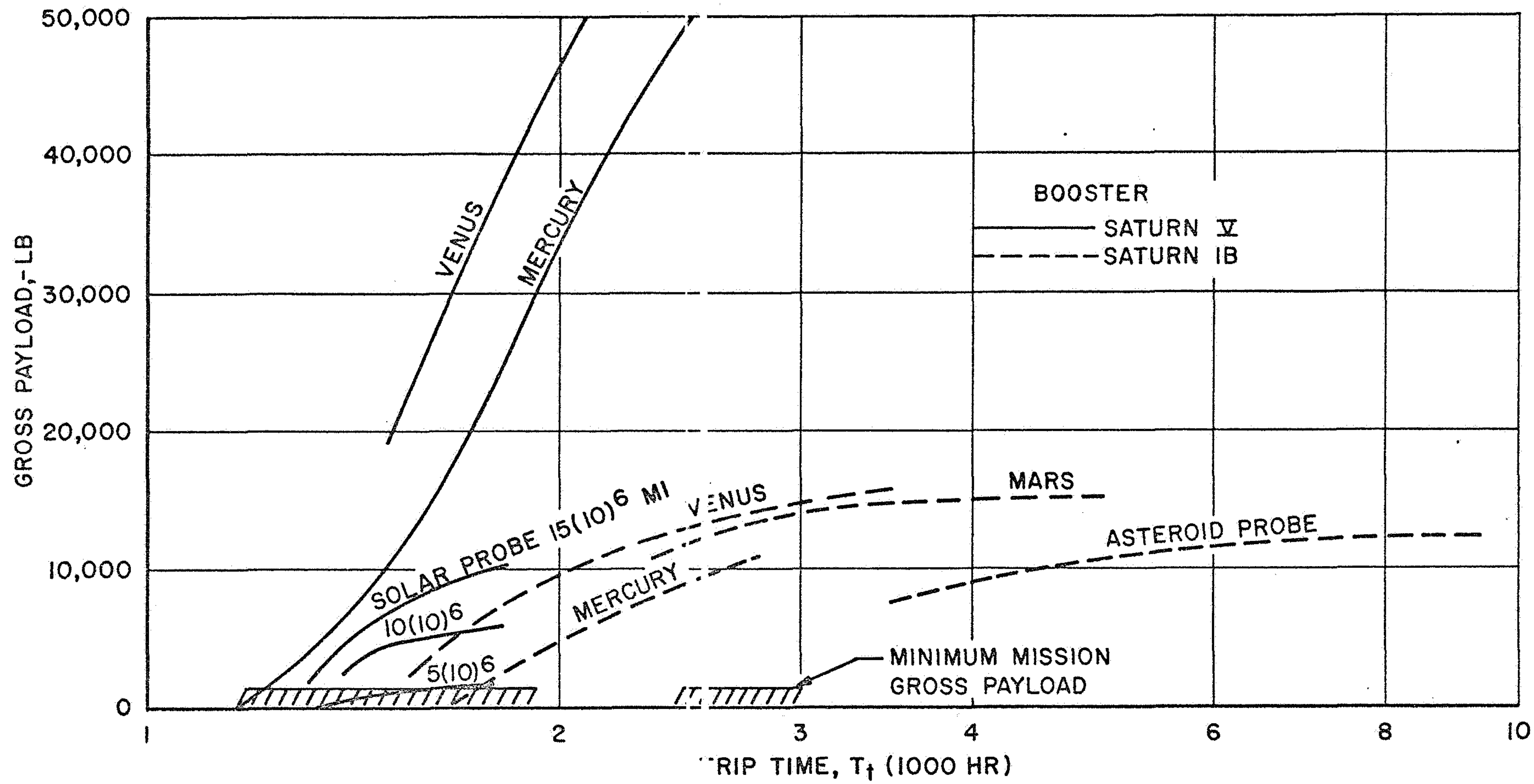


Figure 3.1-3. Probe and Minor Planet Flyby Performance Summary - High Thrust Propulsion Gross Payload

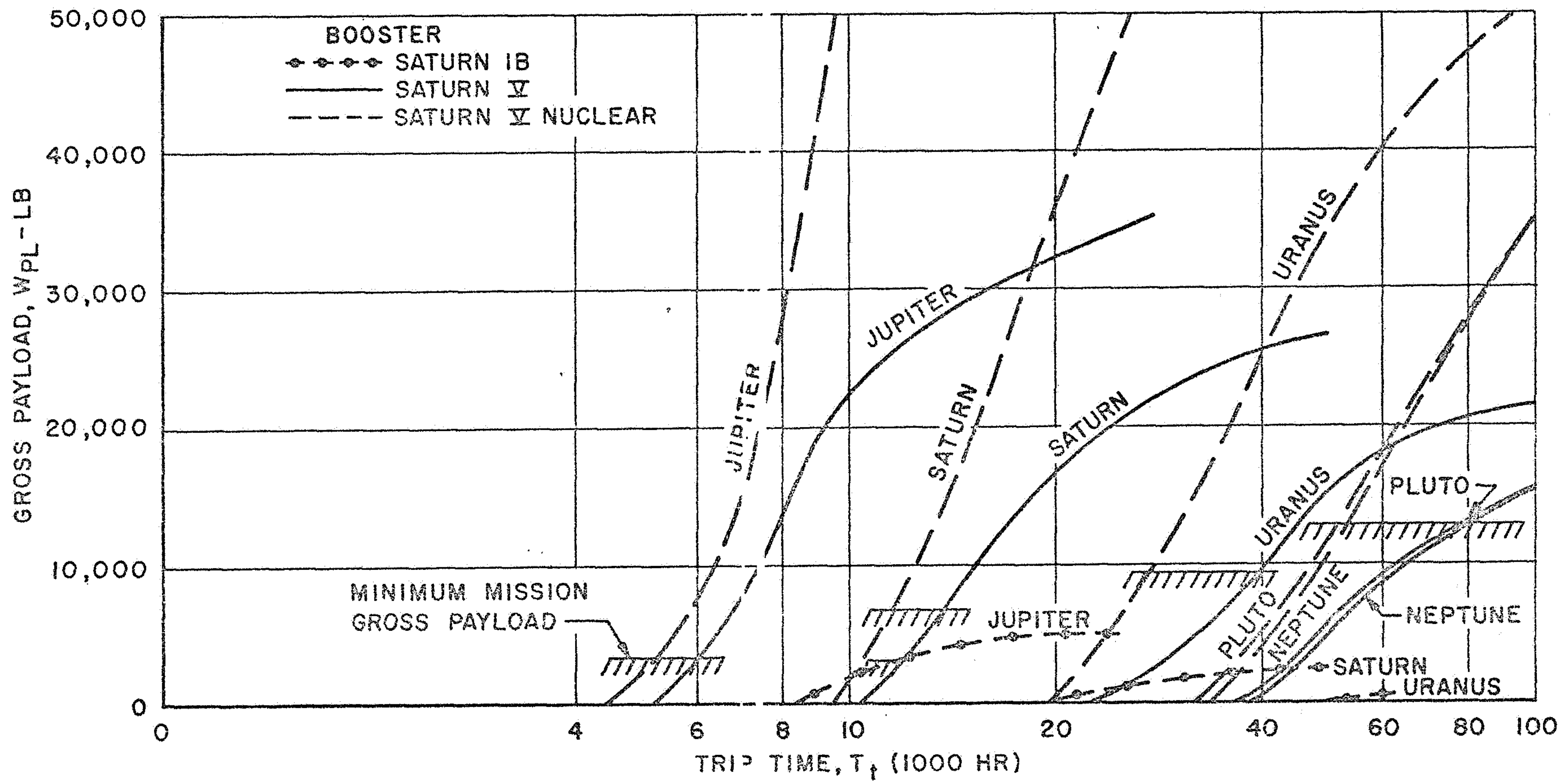


Figure 3.1-4. High Thrus: Propulsion Gross Payload

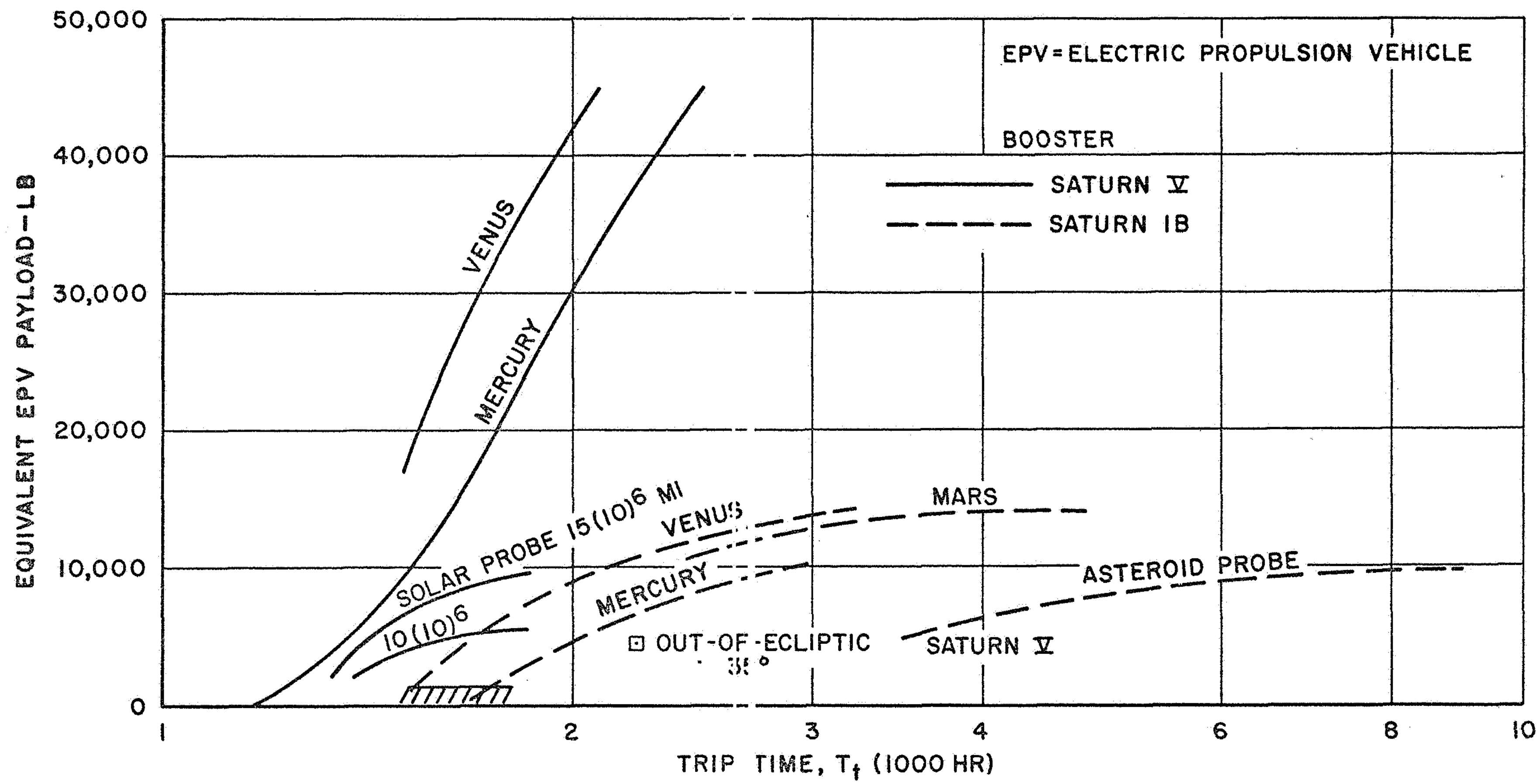


Figure 3.1-5. Probe and Minor Planet Flyby Performance Summary - High Thrust Propulsion Equivalent EPV Payload

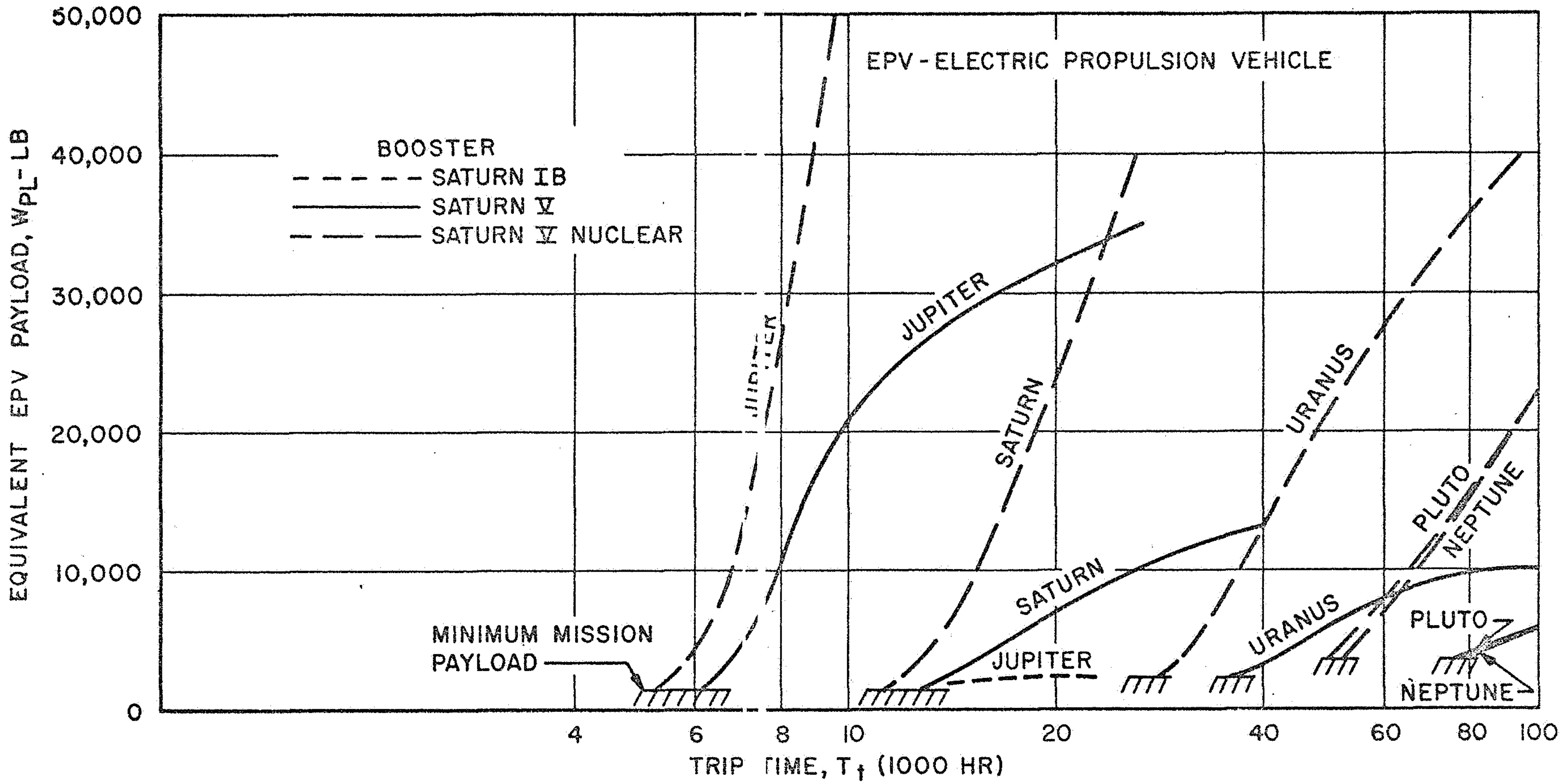


Figure 3.1-6. Major Planet Flyby Performance Summary - High Thrust Propulsion Equivalent EPV Payload

The data produced by the short-lived atmospheric probe can be stored aboard the vehicle and transmitted to earth at a convenient rate; however, the 165 pound "Lander" instrument package will produce information continuously at a rate of about 10^3 bits/sec. This information plus the 2×10^3 bits/sec produced by the 480 pounds of vehicle scientific instruments, will require a minimum communications rate of 36,000 bits/sec. This is based upon a single transmittal of the data over a 2 hour period per day.

The resulting minimum payload weights for the several "Orbiter" missions are given in Table 3.1-3. The disparity is small for the near planets; however, the disparity is large for the distant planets at even these modest communications rates because of the significant power requirements.

The minimum payloads combine with the gross Non-EPV gross payload capabilities to define an operational exclusion area as shown in Figures 3.1-7 and 3.1-8. The minimum payload and the total vehicle ΔV is greater for the "Orbiter" missions than for the "Flyby" missions and, consequently, the Saturn V booster with all chemical or one nuclear stage is required for all of the missions. The all chemical booster is adequate for the minor planet missions; however a nuclear rocket stage is required for missions to Jupiter and beyond. The booster with a nuclear rocket stage is adequate for the Jupiter I and Saturn I missions; however, all the other orbiter missions, including the low altitude Jupiter and Saturn mission, and the Uranus, Neptune, and Pluto missions, are beyond the capability of the chemical plus neclear rocket system.

The Non-EPV gross payloads are related to the equivalent EPV payloads as shown in Figure 3.1-9. These relationships, can be used to equate the gross Non-EPV payloads given in Volume 1 to equivalent EPV payloads with the results shown in Figures 3.1-10 and 3.1-11. These payloads may be compared directly with those given for the electric propulsion vehicles.

Table 3.1-3. Minimum Pay[illegible]
Weight, [illegible]

Planet	Communications Power, KWe	Scientific Instruments	Guidance & Stabilization	Communication Antenna	Communi[illegible] Transm[illegible]
Mercury, Venus, Mars	0.15	3,655	500	350	[illegible]
Jupiter	4.5	3,655	500	350	[illegible]
Saturn	15	3,655	500	350	2[illegible]
Uranus	45	3,655	500	350	67[illegible]
Neptune, Pluto	150	3,655	500	350	2,2[illegible]

oads for Orbiter Vehicles
s.

ation tter	High Temp. Cooling	Power System	Structure	Non-Electric Propulsion Vehicle Payload Weight	Equivalent Electric Propulsion Vehicle Payload Weight
0	15	50	500	5,100	5,050
5	25	1,900	725	7,220	5,100
5	90	5,200	1,130	11,150	5,350
5	270	7,200	1,430	14,100	6,060
)	900	8,900	1,845	18,400	8,500

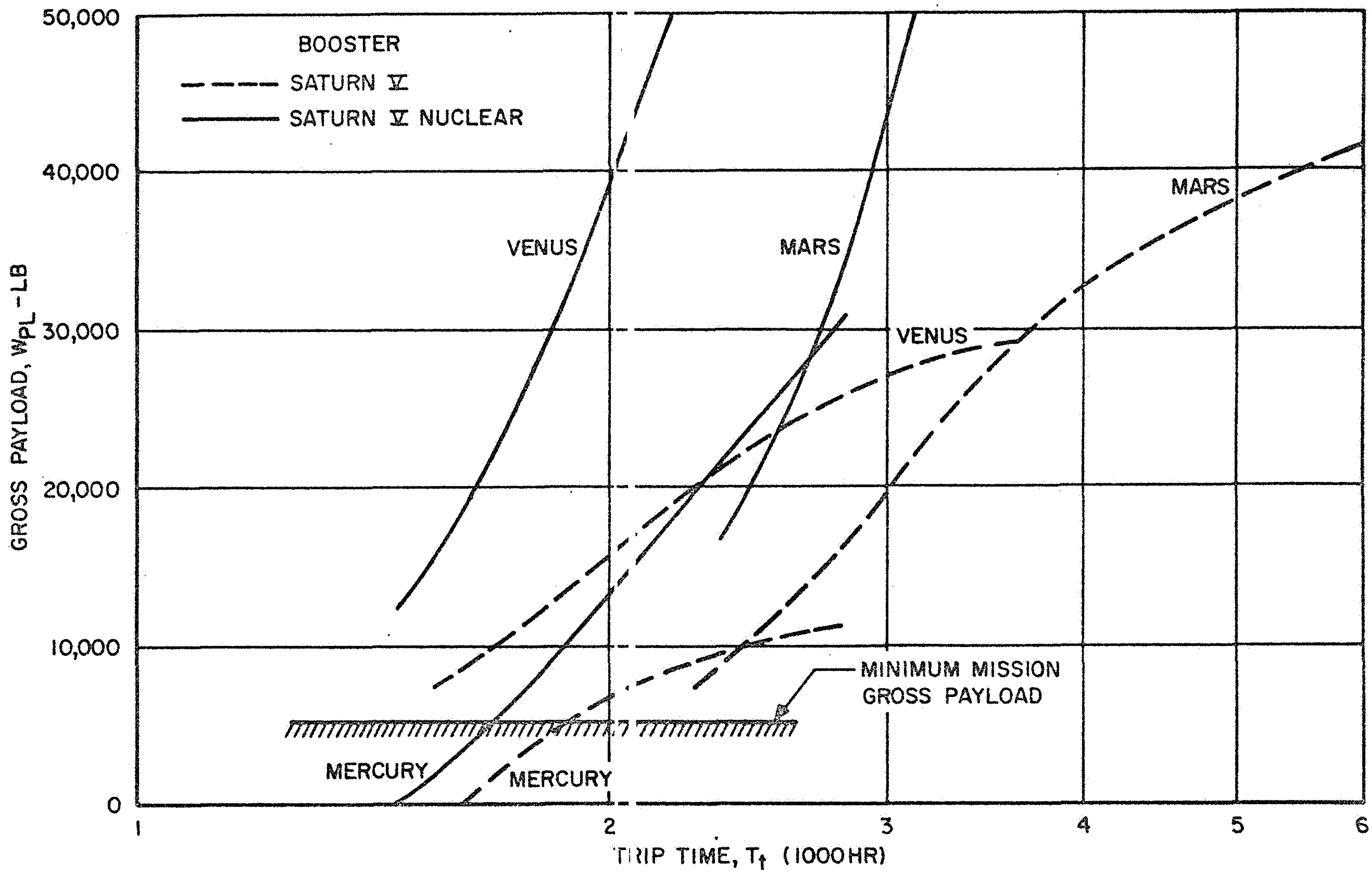


Figure 3.1-7. Minor Planet Orbiter Performance Summary - High Thrust Propulsion

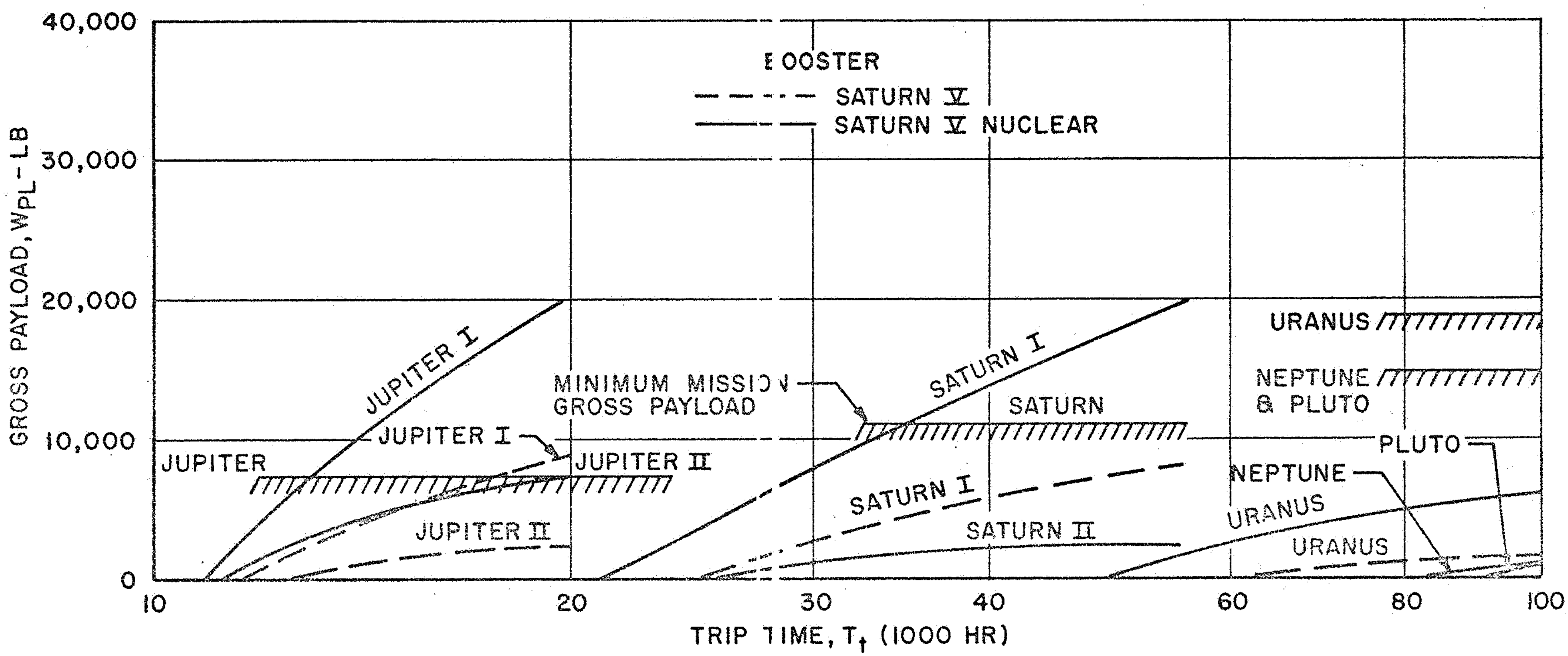


Figure 3.1-8. Major Planet Orbiter Performance Summary - High Thrust Propulsion Gross Payload

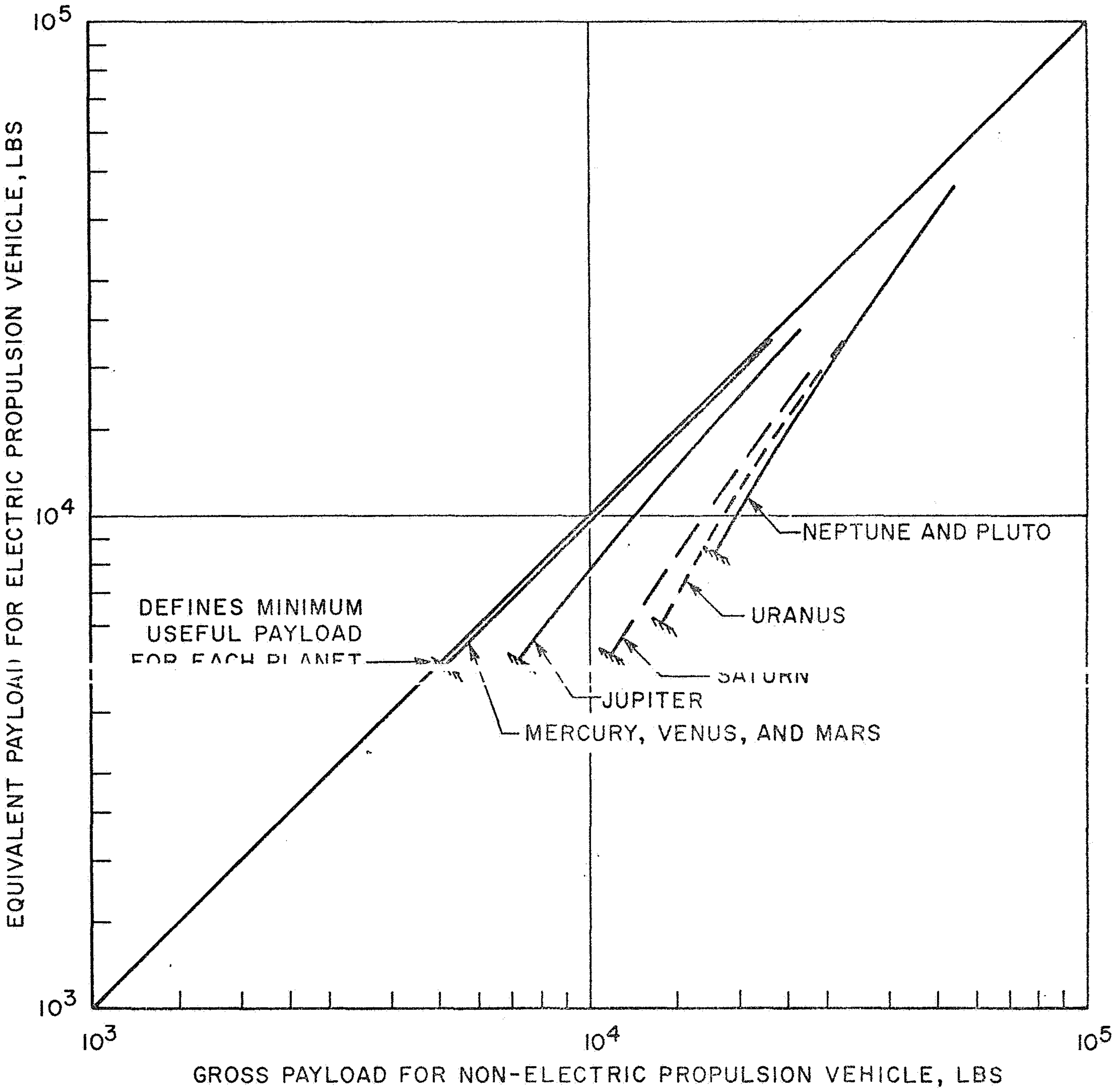


Figure 3.1-9. Relation of Gross and Equivalent Payloads for Orbiter Missions

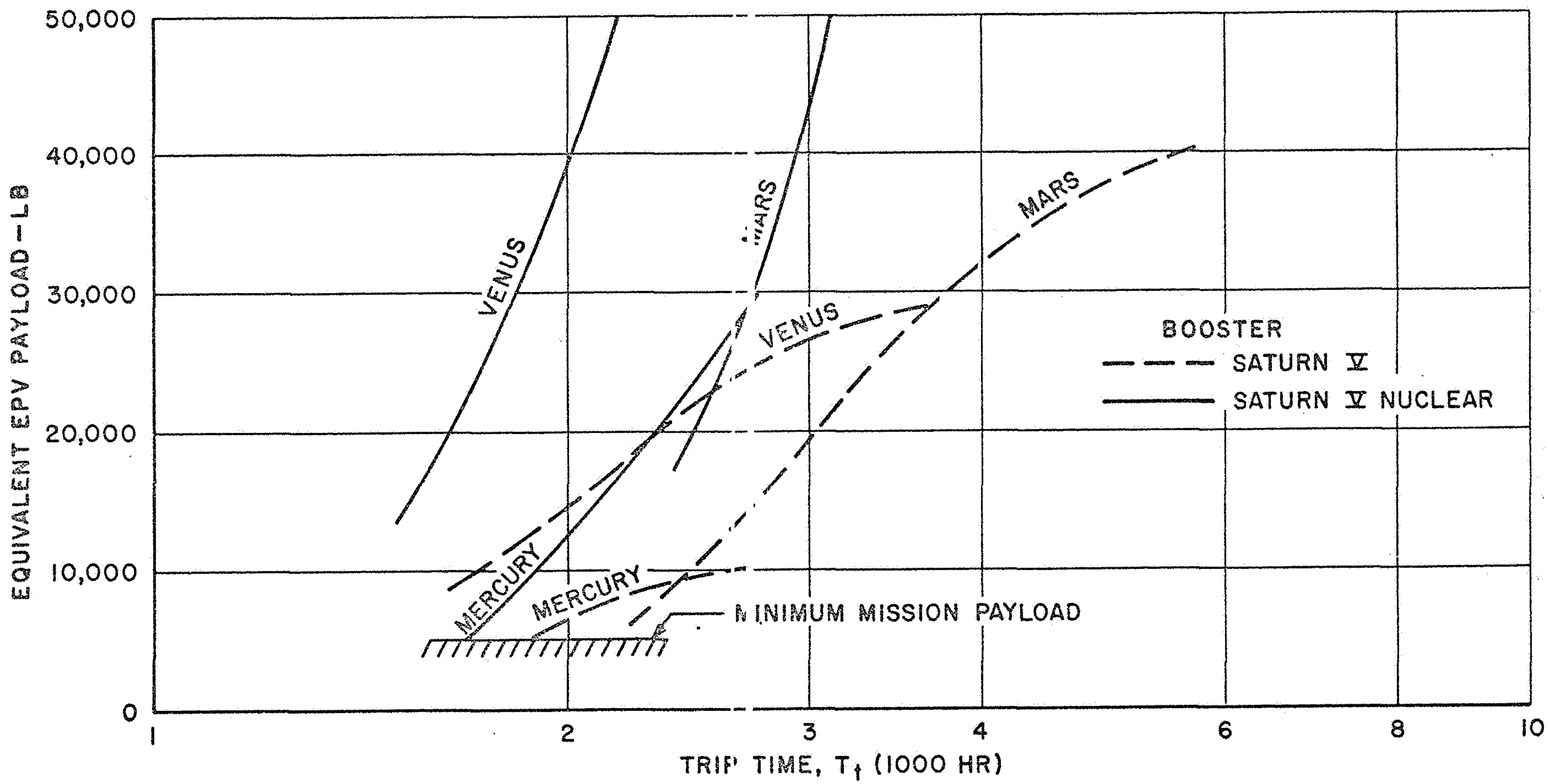


Figure 3.1-10. Minor Planet Orbiter Performance Summary - High Thrust Propulsion Equivalent EPV Payload

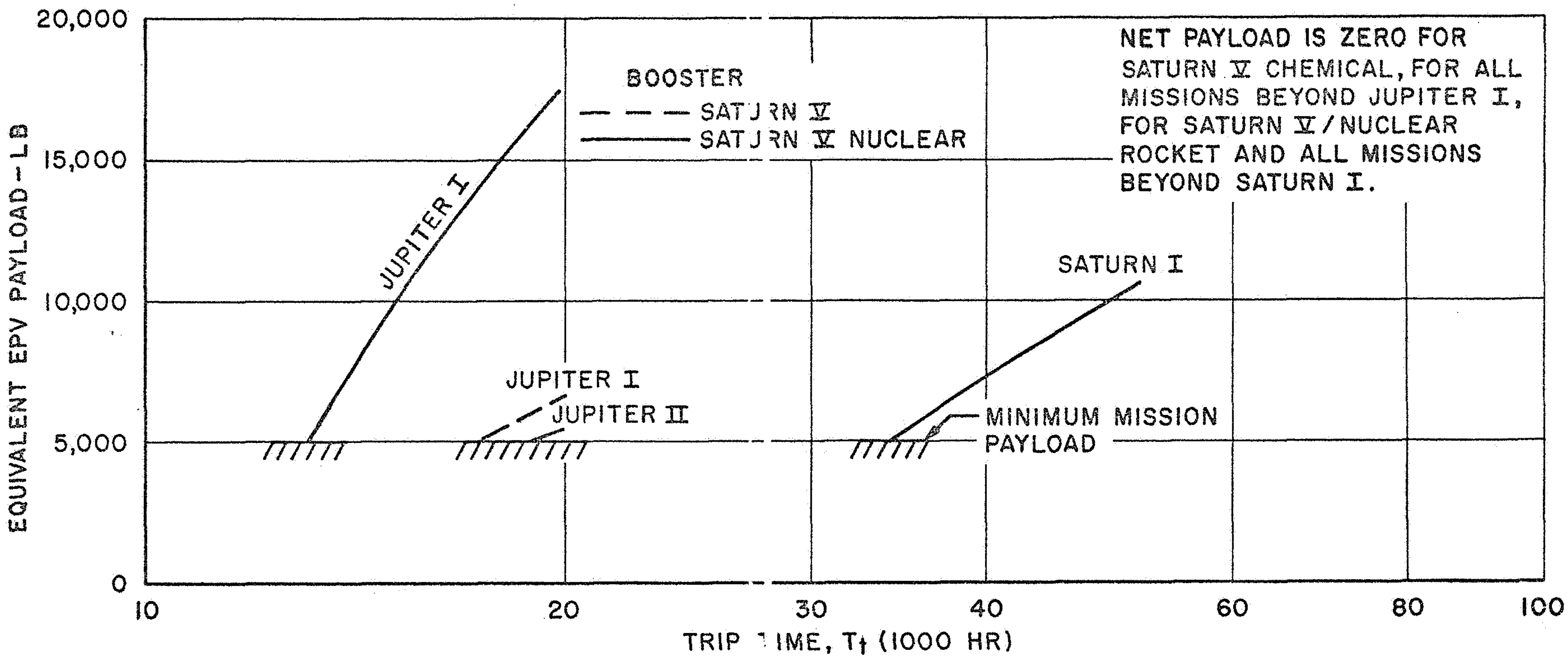


Figure 3.1-11. Major Planet Orbiter Performance Summary - High Thrust Propulsion Equivalent EPV Payload

3.2 INTEGRATION OF MISSION REQUIREMENTS AND POWER SYSTEM CAPABILITIES

Volume 1 defines a wide range of applicable missions for electric propulsion vehicles. However, only a part of that range is attainable because of the limitations on the minimum powerplant specific weights that can be provided at the power levels required by the vehicles.

Of the systems compared in Volume 2 (Section 3.8) it is concluded that the Potassium/Rankine Power System is nearest optimum for the NAVIGATOR missions. These powerplants are expected to result in specific weights in the range of 25 to 70 lb/KWe, depending upon the power level and the state of the technology. The specific weights are given in Figure 3.2-1.

The data on expected specific weights can be combined with the mission requirements to determine the EPV performance capabilities. This is illustrated as follows with the Jupiter II Orbiter performance as an example. Figures 6.2-32 and 6.2-31, Volume 1 are reproduced as Figures 3.2-2 and 3.2-3. The specific-weight/power-level relationships shown on Figure 3.2-1 are cross-plotted on Figure 3.2-2. The operational limits defined by the "Early" and "Improved" technology powerplants are transferred to Figure 3.2-3 by noting the intercept of the technology line with the specific weight lines. The two technology lines on Figure 3.2-3 then define the maximum payload that may be delivered in a given trip time. The area to the right of each technology line defines the operational region of significance. However, all of this range is not available because of the minimum payload limitation of 1500 pounds which is also shown on Figure 3.2-3.

In addition, for comparison, the equivalent payload for a Saturn V vehicle with a nuclear rocket stage is plotted on Figure 3.2-3. As shown, the payload is only slightly greater than the estimated minimum that will be required. This complex of relationships is shown on Figure 3.2-4 with the extraneous data deleted for clarity and with only the allowable operational areas shown. It is clear in the case shown, that the non-electrical propulsion vehicle is marginal in performance and that electric propulsion will be required to deliver a useful payload.

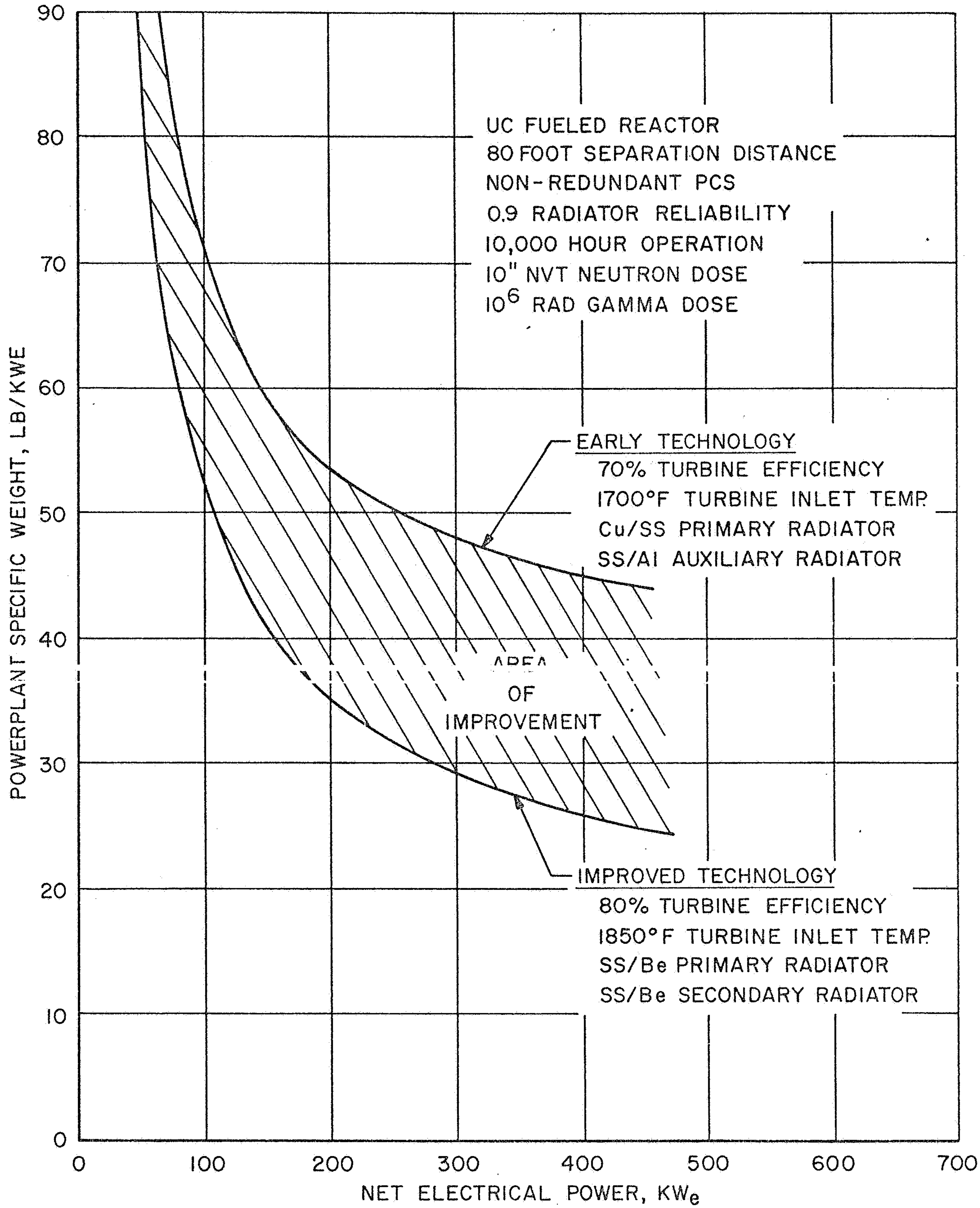


Figure 3.2-1. Recommended Powerplant Specific Weight - Potassium/Rankine Power System

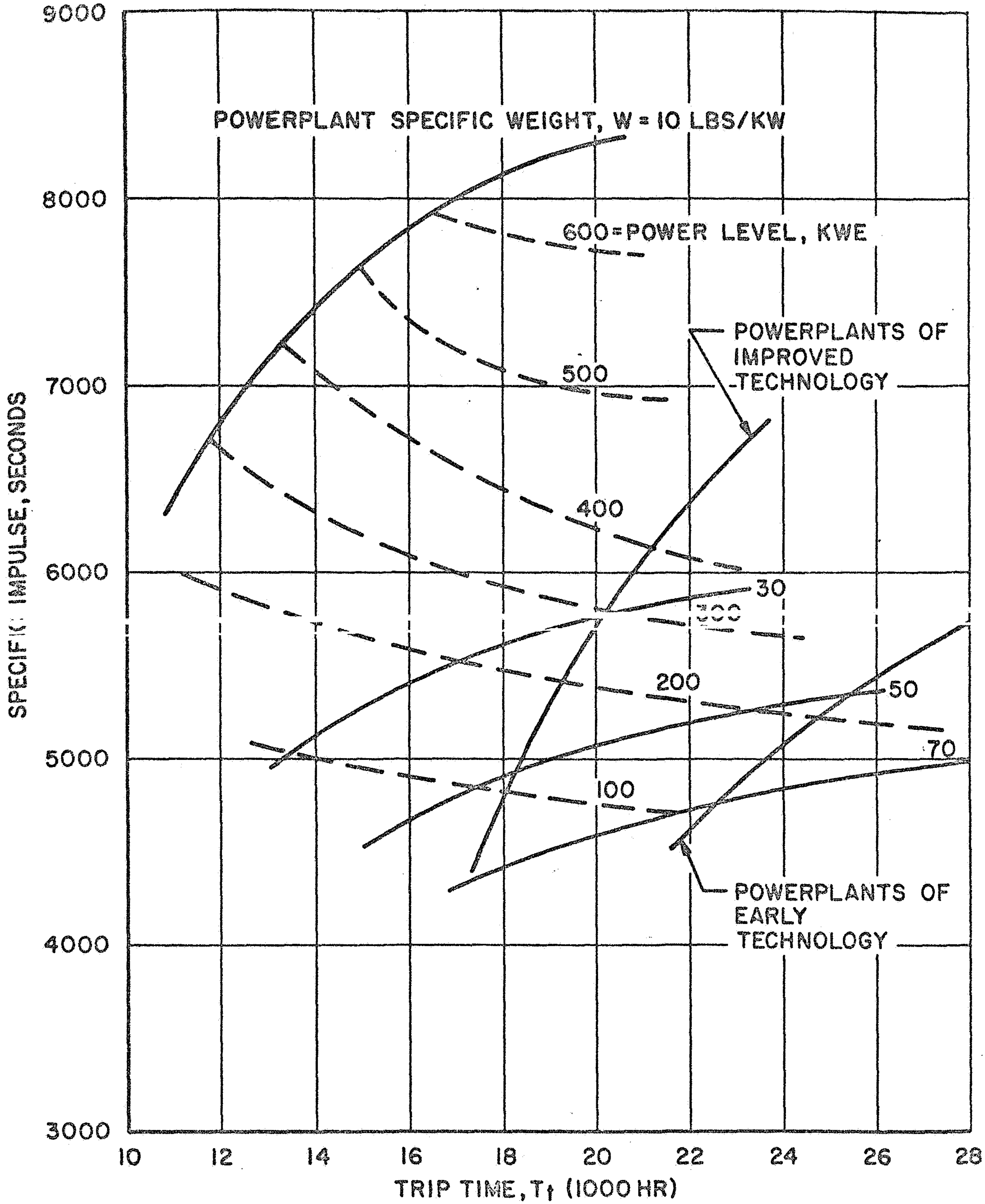


Figure 3.2-2. Jupiter II Orbiter Requirements

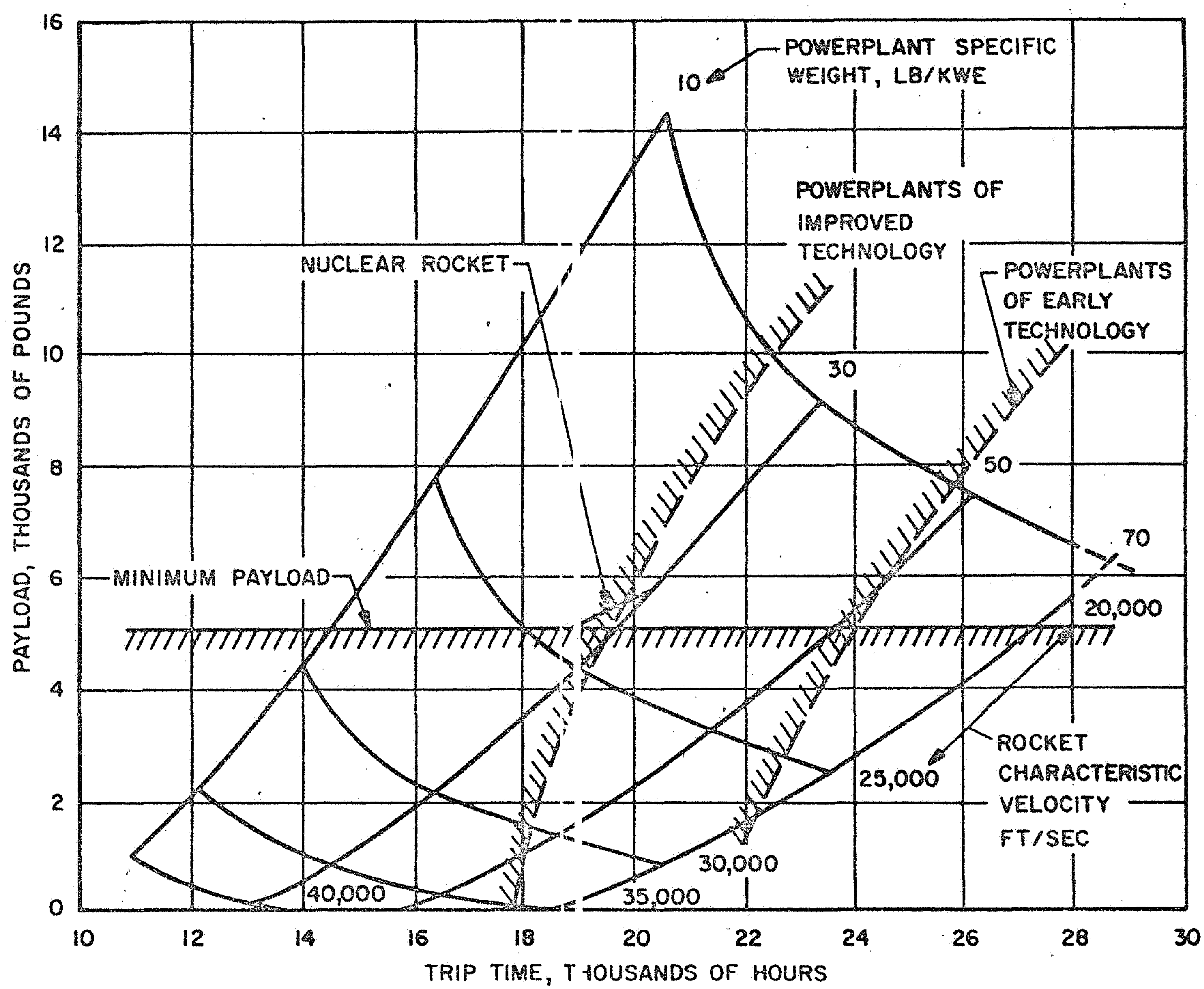


Figure 3.2-3. Comparison of Jupiter II Orbiter Performance Capabilities

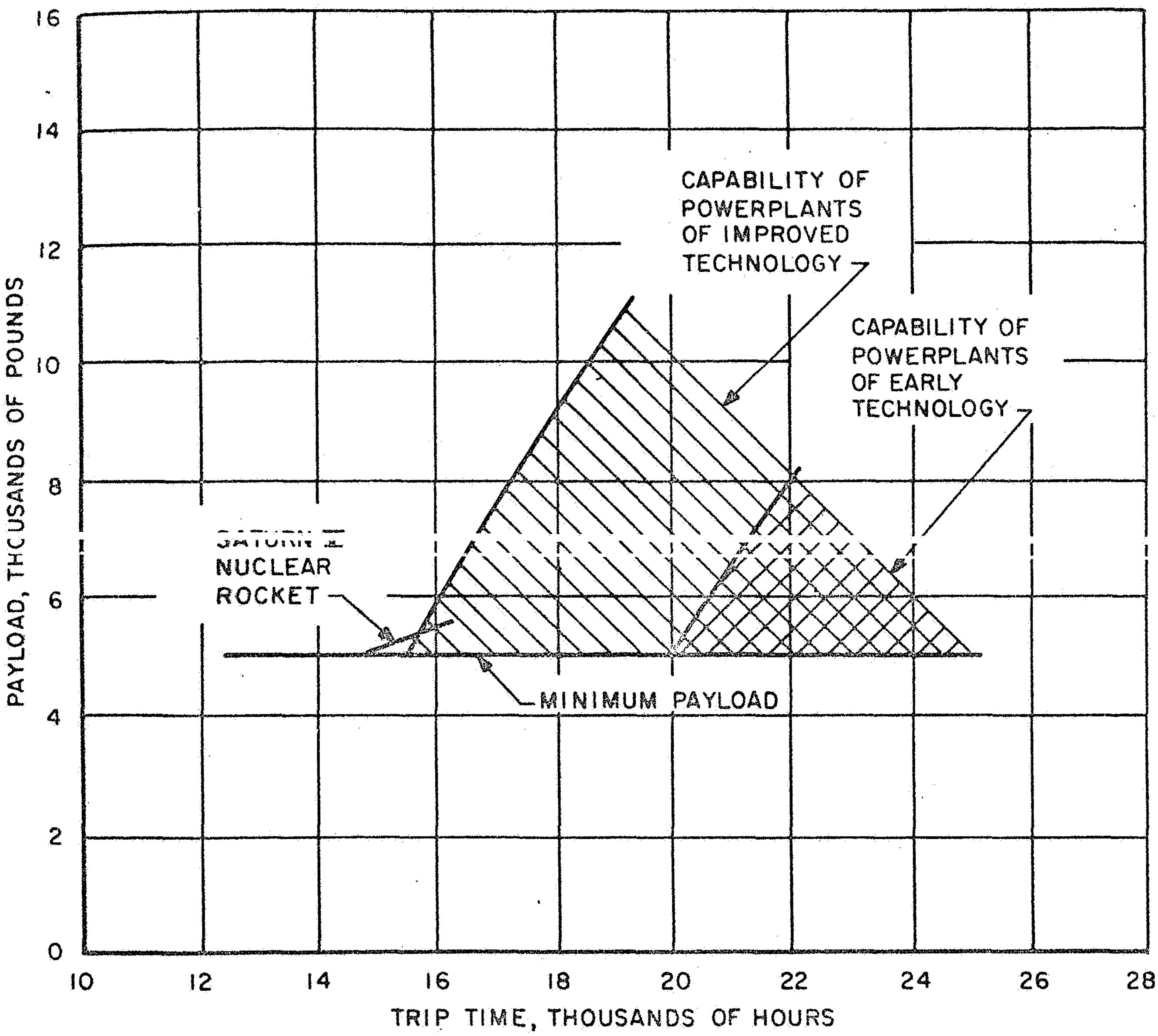


Figure 3.2-4. Comparison of Jupiter II Orbiter Performance Capabilities

The integrated performance curve may be used to determine the performance of powerplants that are not necessarily optimum for the particular mission (i.e., a powerplant that is not exactly the power level and specific weight specified for the optimum mission). Each point on the optimum performance line represents a discrete power level and specific weight that can be obtained for each mission from the proper figure in Volume I. If excess payload is available, the extra weight can be used as additional powerplant weight, thus allowing a specific weight greater than the optimum. For example, Figure 3.2-2 shows that a power level of 440 KW and a specific weight of 25 lb/KWe provide maximum payload at 22,000 hour trip time with an improved technology powerplant.

The corresponding payload from Figure 3.2-3 is 10,000 pounds and the powerplant weight is 11,000 pounds. The payload weight may be decreased and the powerplant weight and specific weight increased as shown in Table 3.2-1.

Table 3.2-1. Selection of Non-Optimum Powerplants for Jupiter II Orbiter Mission

Trip Time hrs.	Power Level KWe	Payload lb.	Payload Alotted for Powerplant Excess Weight lb.	Powerplant Specific Weight lb/KWe
22,000	440	10,000	0	25
22,000	440	7,800	2,200	30
22,000	440	5,600	4,400	35
22,000	440	3,400	6,600	40
22,000	440	1,200	8,800	45

3.3 COMPARISON OF VEHICLE PROPULSION SYSTEMS

Integrated performance capability curves similar to that described above were prepared for each of the NAVIGATOR "Flyby" and "Orbiter" missions defined in Table 3.3-1 with the results shown in Figures 3.3-1 and 3.3-2.

Table 3.3-1. NAVIGATOR Mission Summary

Mission Type	Mission	Terminal Condition
Fly-by	Solar Probe	5 $(10)^6$ Miles
	Mercury	Optimum Fly-By
	Asteroid Belt	Optimum Fly-By
	Jupiter	Optimum Fly-By
	Saturn	Optimum Fly-By
	Uranus	1975 Fly-By
	Neptune	1986 Fly-By
	Pluto	1986 Fly-By
	Out-of-the-Ecliptic	35 Degrees
Orbiter	Mercury	2,000 Miles Radius
	Venus	5,000 Miles Radius
	Mars	3,000 Miles Radius
	Jupiter I	1,170,000 Miles Radius
	Jupiter II.	262,000 Miles Radius
	Saturn I	760,000 Miles Radius
	Saturn II	44,000 Miles Radius
	Uranus	20,000 Miles Radius
	Neptune	20,000 Miles Radius
	Pluto	5,000 Miles Radius

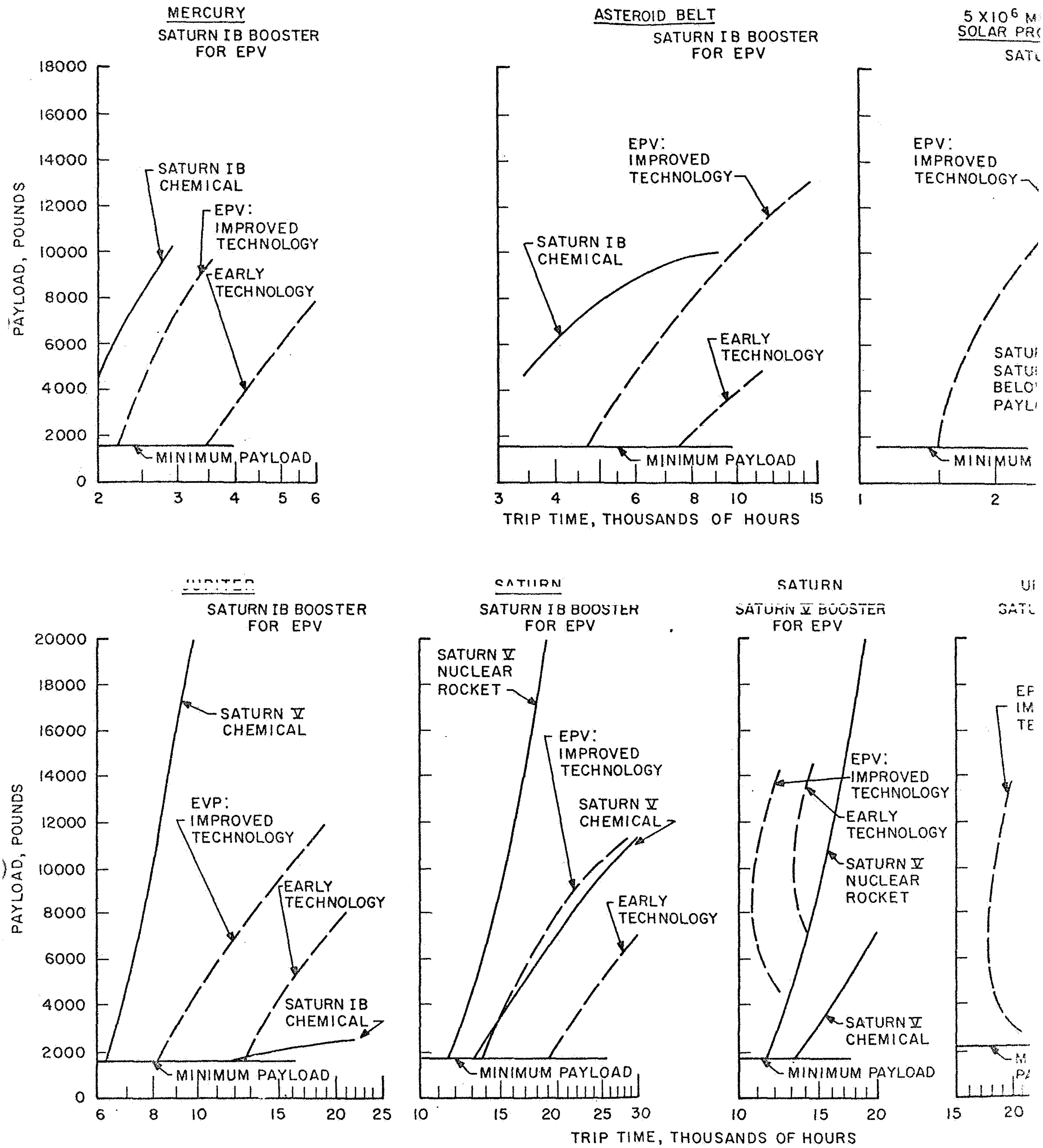


Figure 3.3-1. Summary C
All Chemical, a

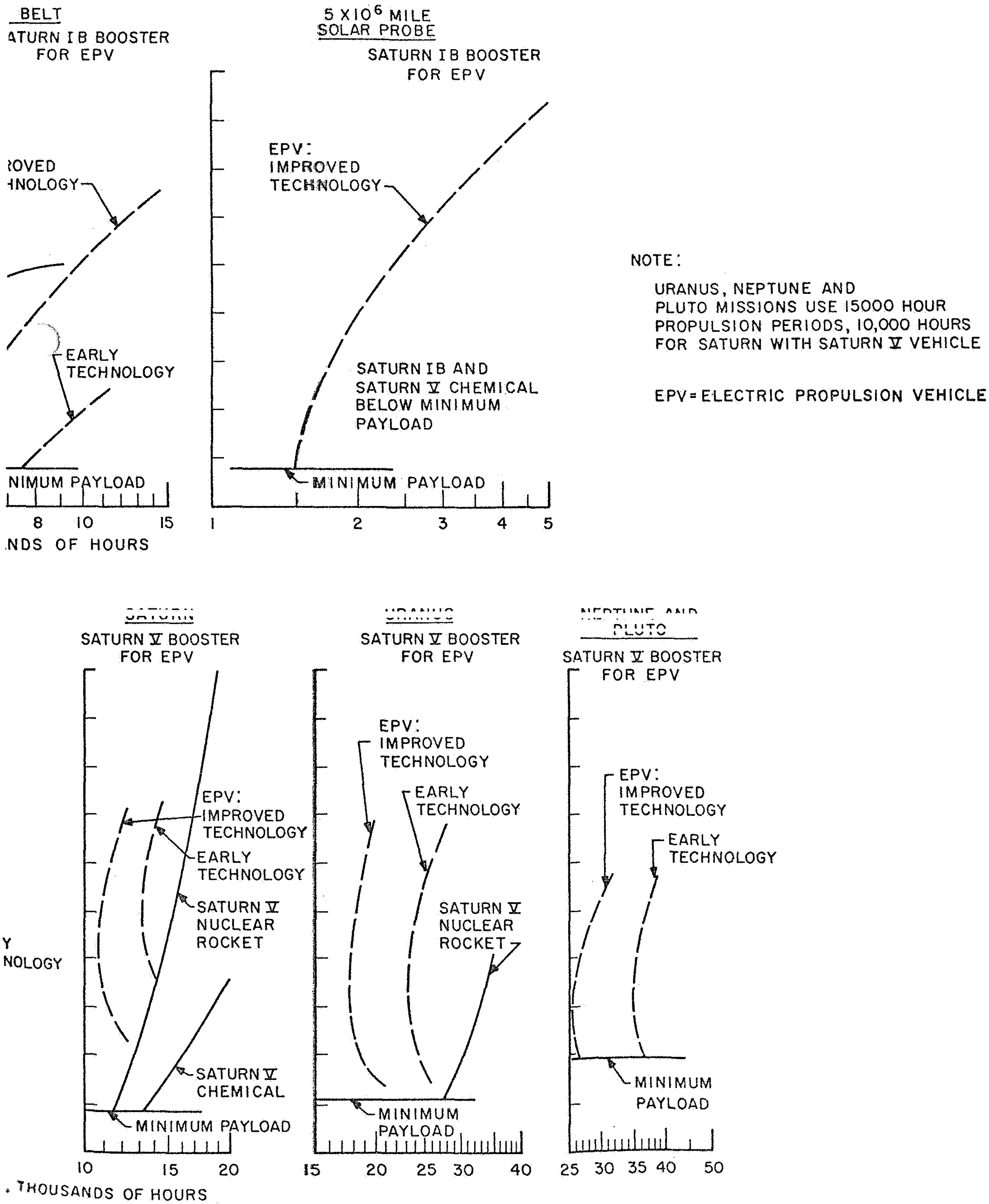


Figure 3.3-1. Summary Comparison of the Flyby Payload Capabilities of Nuclear Electric, All Chemical, and Chemical Plus Nuclear Rocket Propulsion Vehicle

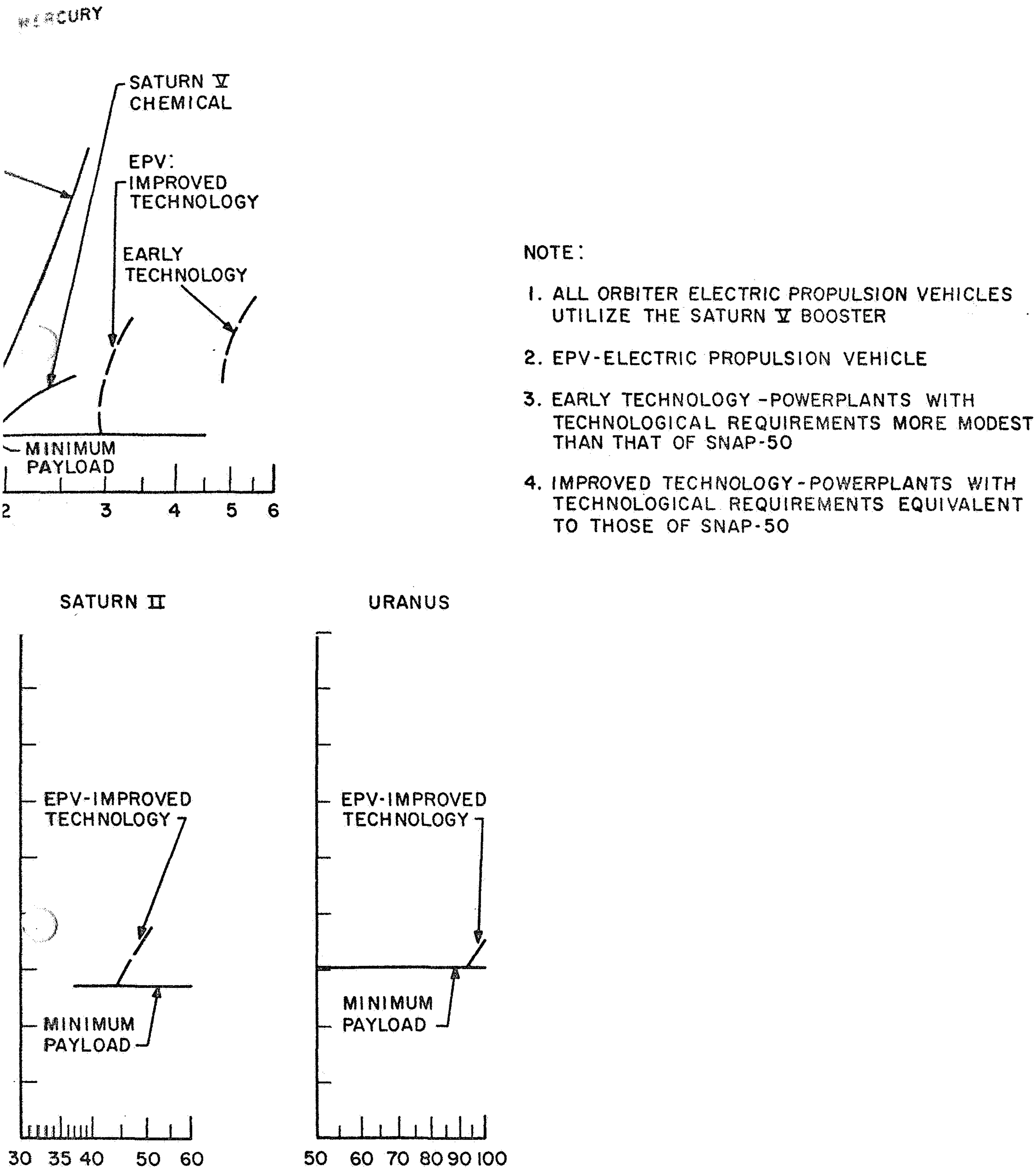
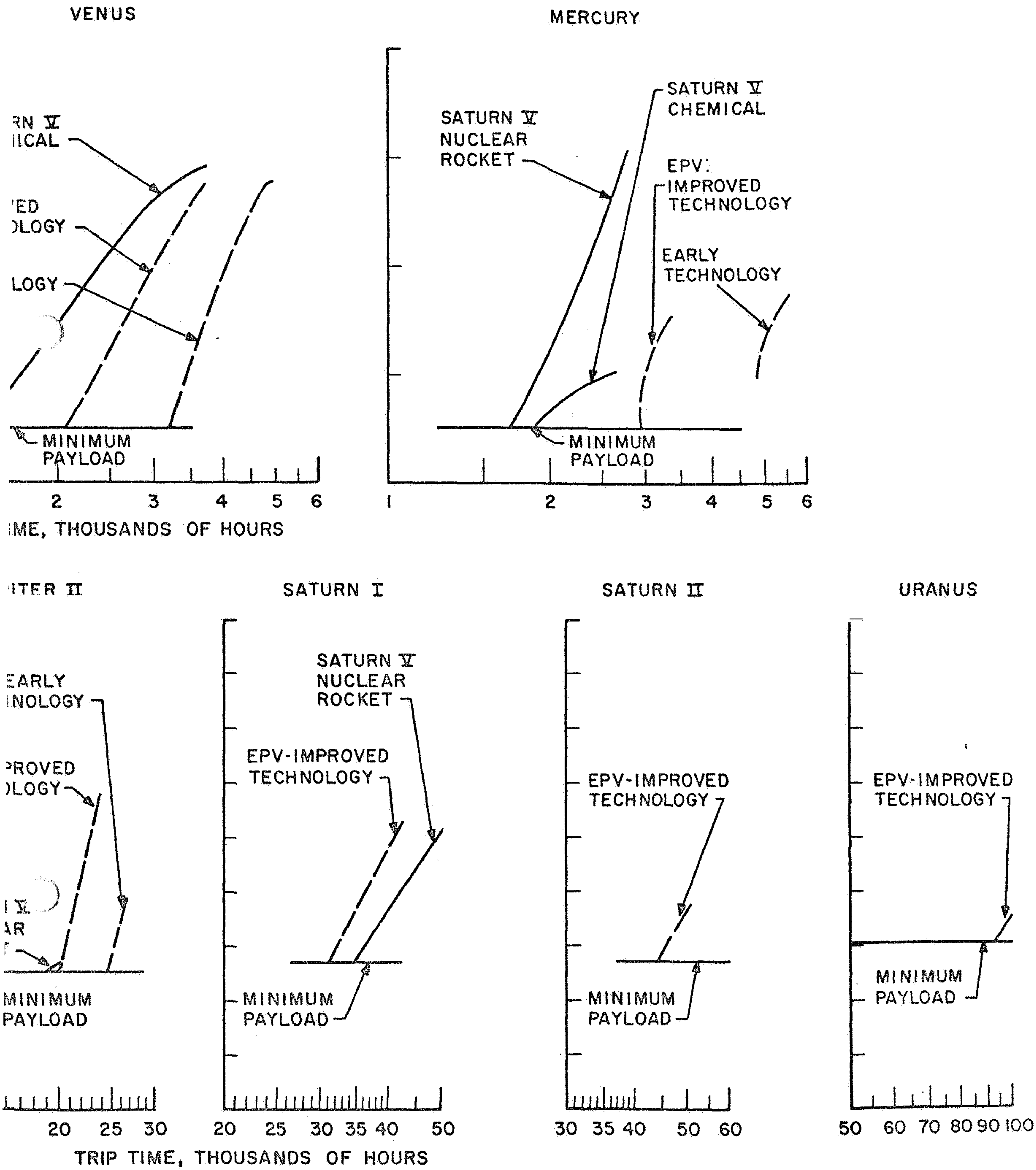


Figure 3.3-2. Summary Comparison of the Orbiter Payload Capabilities of Nuclear Electric, All Chemical, and Chemical Plus Nuclear Rocket Propulsion Vehicles



NOTE:

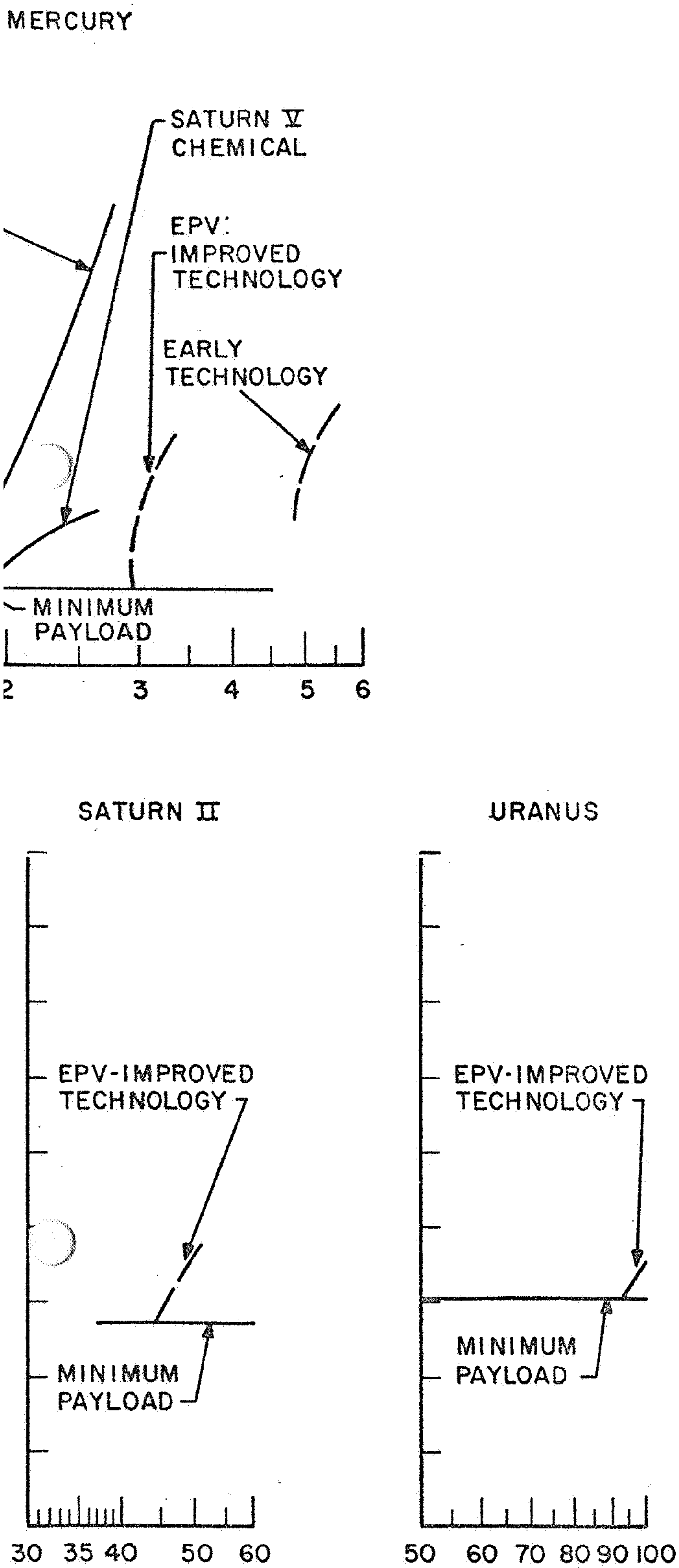
1. ALL C
UTIL

2. EPV-

3. EARL
TECH
THAN

4. IMPR
TECH
TO T

Figure 3.3-2. Summary Comparison of the Orbite
All Chemical, and Chemical Plus Nucle



NOTE:

1. ALL ORBITER ELECTRIC PROPULSION VEHICLES UTILIZE THE SATURN V BOOSTER
2. EPV-ELECTRIC PROPULSION VEHICLE
3. EARLY TECHNOLOGY-POWERPLANTS WITH TECHNOLOGICAL REQUIREMENTS MORE MODEST THAN THAT OF SNAP-50
4. IMPROVED TECHNOLOGY-POWERPLANTS WITH TECHNOLOGICAL REQUIREMENTS EQUIVALENT TO THOSE OF SNAP-50

Figure 3.3-2. Summary Comparison of the Orbiter Payload Capabilities of Nuclear Electric, All Chemical, and Chemical Plus Nuclear Rocket Propulsion Vehicles

3.3.1 FLYBY VEHICLE PERFORMANCE (Figure 3.3-1)

The Mercury Flyby can be accomplished with the Saturn IB Booster with either chemical or nuclear electric propulsion. The power range for the electric systems is generally below 300 KWe and the propulsion time range is below 5000 hours which is well within the capability of an early nuclear electric power system. However, large payloads can be delivered in less time by the chemical system.

The Asteroid-Belt Flyby Probe can also be accomplished with the Saturn IB. Either the "Early" or "Improved" technology power systems are adequate; however, the all chemical propulsion system is superior.

The Saturn IB is also adequate for the Jupiter mission provided that an electric propulsion system is used. The payload with the SIB all chemical system is clearly marginal and a Saturn V is required to surpass the EPV performance. With the large difference in cost between the S-IB and S-V, there is a strong incentive to utilize the S-IB with an EPV.

The Saturn Flyby mission is similar to that of Jupiter. The EPV can provide the necessary payload with a S-IB instead of a S-V. The Saturn V chemical and Saturn IB EPV performance are essentially equivalent. The EPV with the S-V provides significantly greater capability than either the S-V chemical or S-V Nuclear Rocket propulsion systems.

The All Chemical S-V is not adequate for any mission beyond Saturn. The S-V Nuclear Rocket can provide a large payload for the Uranus mission; however, either of the EPV's will provide significantly superior performance.

The S-V Nuclear Rocket is not adequate for either the Neptune or Pluto missions; however, both missions are well within the capability of the nuclear systems. Operating times of 10,000 or 15,000 hours are adequate and power levels are in the range of 100 to 200KW.

The Saturn IB and V chemical systems are not adequate for the 0.05 a.u. solar probe, although the Saturn V chemical can provide a useful payload at 0.1 a.u. and greater distances.

3.3.2 "ORBITER" VEHICLE PERFORMANCE (Figure 3.3-2)

All of the "Orbiter" missions examined require the use of the Saturn V booster as shown by Figure 3.3-2. The Saturn V chemical system can provide payloads of 20,000 pounds or more for the Mars and Venus missions and a payload of 10,000 pounds for the Mercury mission. Thus, more sophisticated power systems will not be required for the exploration of any of the inner planets.

The S-V chemical system can also provide a modest payload of 6,000 pounds for the Jupiter I Mission; however, the S-V Nuclear Rocket of the electric propulsion system will likely be required to provide additional payload capability.

For missions beyond Jupiter I, the Saturn V with electric propulsion will be required. The Jupiter II and Saturn I missions are well within the nuclear electric system capabilities; however, for Saturn II and Uranus, the operating time exceeds 20,000 hours.

The Neptune and Pluto missions will require the use of EPV; however, the mission profile assumed in these studies (i.e., launch of the EPV to greater than escape velocity with a S-V plus additional chemical stages) is such that the operating times and trip times are excessive. Also, the optimum mission vehicle requires less than 100 KWe of power and specific weights of 50 lb/KWe cannot be attained at that level.

3.3.3 SUMMARY COMPARISON OF PROPULSION SYSTEM PERFORMANCE

The previous comparison of propulsion systems can be summarized as shown in Tables 3.3-2 and 3.3-3. The tables define:

Table 3.3-2 Comparison of

Mission	Saturn IB Plus Two Chemical Stages		PRO					
			SATURN IB Plus					
			Early Technology Electric Propulsion Vehicle			Improved Technology Electric Propulsion Vehicle		
	Payload Lbs	Trip Time Hrs	Payload Lbs	Trip Time Hrs	Pro-pulsion Time Hrs	Payload Lbs	Trip Time Hrs	P pul Tim
Mercury	$\boxed{5,000}$	$\boxed{2,050}$	5,000	4,600	4,200	5,000	2,600	1,
Asteroid Belt	$\boxed{5,000}$	$\boxed{3,500}$	5,000	11,600	9,200	5,000	6,050	5,
Jupiter	2,200	16,000	$\boxed{5,000}$	$\boxed{16,000}$	$\boxed{11,300}$	$\boxed{5,000}$	$\boxed{10,200}$	$\boxed{6,}$
Saturn			7,000	30,000	21,500	$\boxed{7,000}$	$\boxed{18,500}$	$\boxed{12,}$
Uranus								
Neptune & Pluto					PROPULSION SYSTEMS INADEQUATE			
Solar Probe 5×10^6 mile								

lsion Systems for Navigator Flyby Missions

ION SYSTEMS									
Saturn V Plus Two Chemical Stages		Saturn V Plus One Chemical and One Nuclear Rocket Stage		SATURN V Plus One or Two Chemical Stages AND					
				Early Technology Electric Propulsion Vehicle			Improved Technology Electric Propulsion Vehicle		
Payload Lbs	Trip Time Hrs	Payload Lbs	Trip Time Hrs	Payload Lbs	Trip Time Hrs	Pro-pulsion Time Hrs	Payload Lbs	Trip Time Hrs	Pro-pulsion Time Hrs
				PROPULSION SYSTEMS GREATER THAN REQUIRED					
5,000	7,100								
7,000	19,500	7,000	14,000	7,000	14,000	10,000	7,000	11,000	10,000
		7,000	34,000	7,000	23,000	15,000	7,000	18,000	15,000
				7,000	35,000	15,000	7,000	26,000	15,000
		Not Available		5,100	2,500	2,000	10,800	2,500	2,000

Table 3.3-3 Comparison of Propulsion Systems for Navigator "Orbiter" Missions

Mission	PROPULSION SYSTEMS									
	Saturn V Plus Two Chemical Stages		Saturn V Plus One Chemical and One Nuclear Rocket Stage		Saturn V Plus One or Two Chemical Stages and					
					Early Technology Electric Propulsion Vehicle			Improved Technology Electric Propulsion Vehicle		
	Payload Lbs	Trip Time Hrs	Payload Lbs	Trip Time Hrs	Payload Lbs	Trip Time Hrs	Propulsion Time Hrs	Payload Lbs	Trip Time Hrs	Propulsion Time Hrs
Mars	10,000	2,450	10,000	2,100	1[illegible],000	3,800	3,000	10,000	2,700	1,700
Venus	10,000	1,700	10,000	1,400	1[illegible],000	3,400	2,800	10,000	2,350	1,400
Mercury	10,000	2,550	10,000	1,900	1[illegible],000	4,800	4,000	10,000	3,000	1,500
Jupiter I	7,000	20,500	7,000	14,000	[illegible],000	19,500	8,000	7,000	17,000	6,000
Jupiter II			5,350	14,000	[illegible],350	24,000	12,000	12,000	24,000	8,000
Saturn I			7,300	40,000	[illegible],700	40,000	14,000	9,600	40,000	14,000
Saturn II					[illegible],600	50,000	20,000	7,500	50,000	20,000
Uranus								6,100	93,000	25,000
Neptune										
Pluto										

PROPULSION SYSTEMS INADEQUATE

- The general range of missions in which a particular propulsion system will find application,
- The range in which the systems either exceed or do not meet the minimum payload requirements, and
- The propulsion system that is nearest optimum for the mission.

The systems are compared for discrete missions at either equivalent payload or equivalent trip time. The optimum propulsion system is selected, based upon consideration of payload, booster cost, and system capability. Generally, the Saturn IB or V boosters with additional chemical stages are selected to the limit of their capability. Substitution of either nuclear rocket or electric propulsion vehicles for the 3rd and 4th chemical stages on the S-V will involve additional cost and, therefore, will likely be delayed for missions in which the all chemical system cannot deliver the minimum required payload.

The Mercury, Asteroid Belt, and Jupiter Flyby Missions are compared (Table 3.3-2) at payloads of 5,000 pounds. The SIB with added chemical stages is adequate for the Mercury and Asteroid Missions as indicated. The Jupiter mission can be accomplished with electric propulsion with the saving of cost between a S-V and a S-IB.

The Saturn through Pluto Flyby Missions are compared at a payload of 7,000 pounds. As in the case of the Jupiter Mission, the Saturn mission can be accomplished with the S-IB with electric propulsion whereas, the chemical or nuclear rocket upper stages will require the S-V.

For Uranus, Neptune, and Pluto the electric propulsion system is required and it provides a significantly lower trip time. It is important that the Jupiter, Uranus, Nepturn and Pluto missions can utilize the "early" technology powerplant, thus allowing the orderly development of the "improved" technology powerplant concurrent with the accomplishment of useful missions.

The Mars, Venus, and Mercury "Orbiter" Missions are compared at a payload of 10,000 pounds (Table 3.3-3). The Saturn V chemical propulsion system is adequate for all three missions.

The Jupiter I Mission requires either the S-V chemical or Nuclear Rocket. The Nuclear Rocket upper stage will deliver the same payload in a lower trip time (6500 hours less).

The Jupiter II mission is beyond the capability of the S-V chemical and either Nuclear Rocket or electric propulsion is required. The Nuclear Rocket provides a significantly lower trip time; however, Figure 3.3-2 indicated that 5,600 pounds is the maximum payload that can be delivered. This is near minimal and, therefore, the electric propulsion vehicle will likely be required.

The electric propulsion vehicle provides greater payload for the Saturn I mission and is required for all missions beyond the Saturn I Orbiter.

Tables 3.3-2 and 3.3-3 are repeated as Tables 3.3-4 and 3.3-5 with the comparison details omitted to emphasize the correlation of mission and propulsion system. The tables show that:

- The S-IB with added chemical stages is clearly adequate for 2 Flyby missions (Mercury and Asteroid Belt).

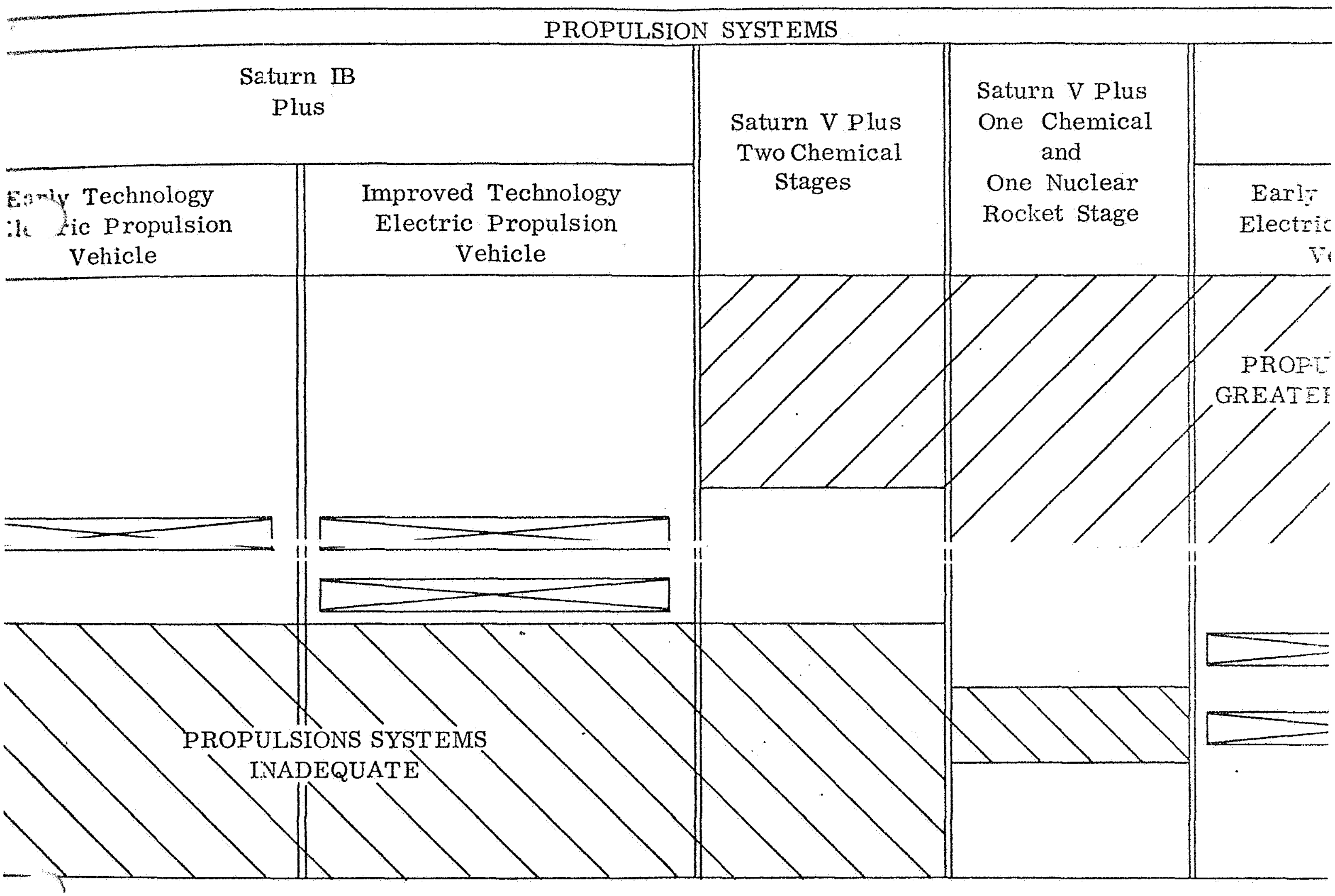
- The S-V with added chemical stages is clearly adequate for 3 Orbiter missions (Mars, Venus, Mercury) and possibly the Jupiter I mission. It is not superior for any of the flyby missions.

- The S-V with a nuclear rocket stage is not required for any of the Flyby missions, but has possible applications in two of the Orbiter Missions.

Table 3.3-4 Comparison of Propuls

MISSION	PROPULS		
	Saturn IB Plus Two Chemical Stages	Saturn IB Plus	
		Early Technology Electric Propulsion Vehicle	Improved Technology Electric Propulsion Vehicle
Mercury			
Asteroid Belt			
Jupiter			
Saturn			
Uranus			
Neptine & Pluto		PROPULSIONS SYSTEMS INADEQUATE	
Solar Probe 5×10^6 mile			

Table 3.3-4 Comparison of Propulsion Systems for Navigator Flyby Missions



stems for Navigator Flyby Missions

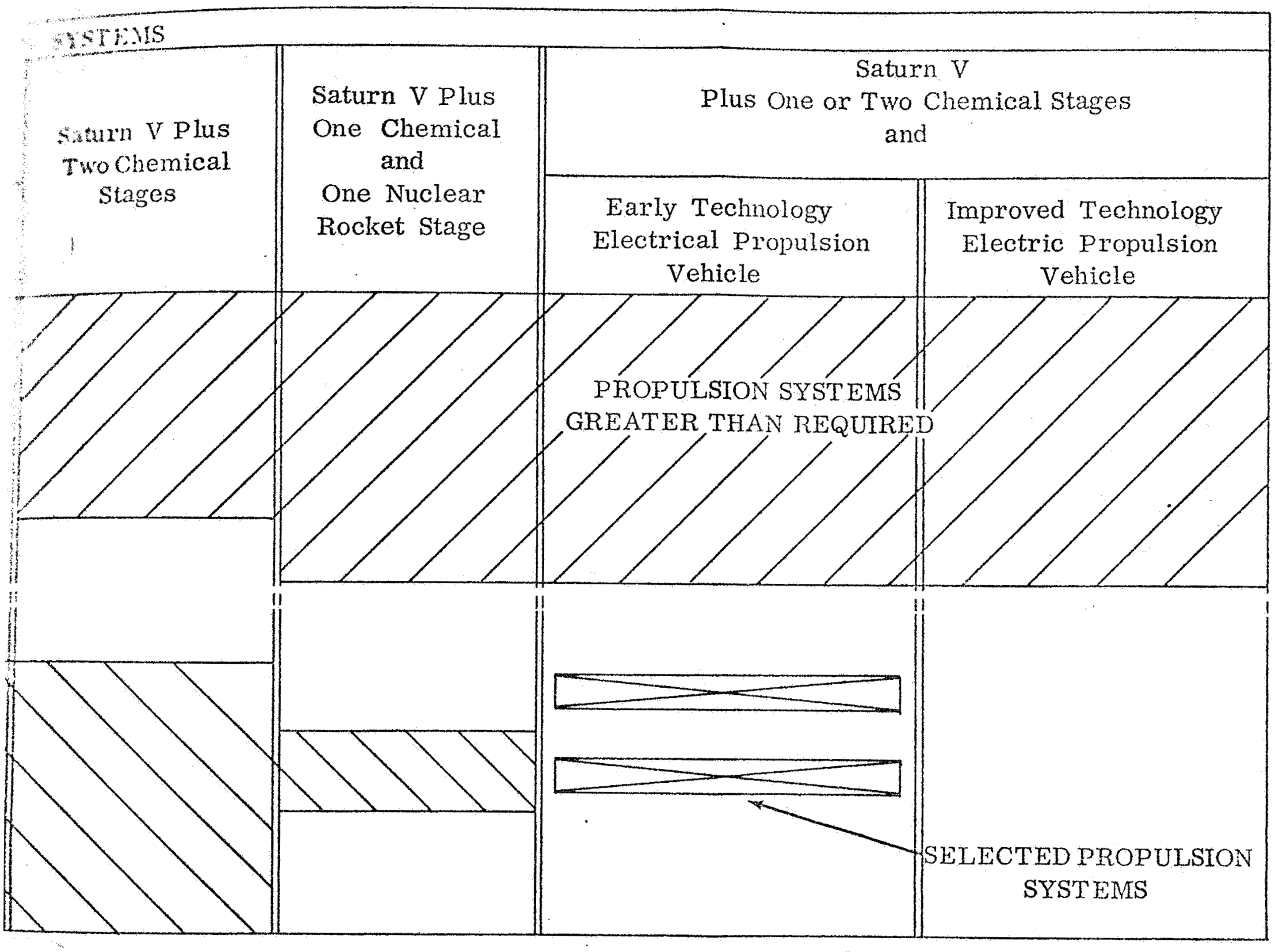
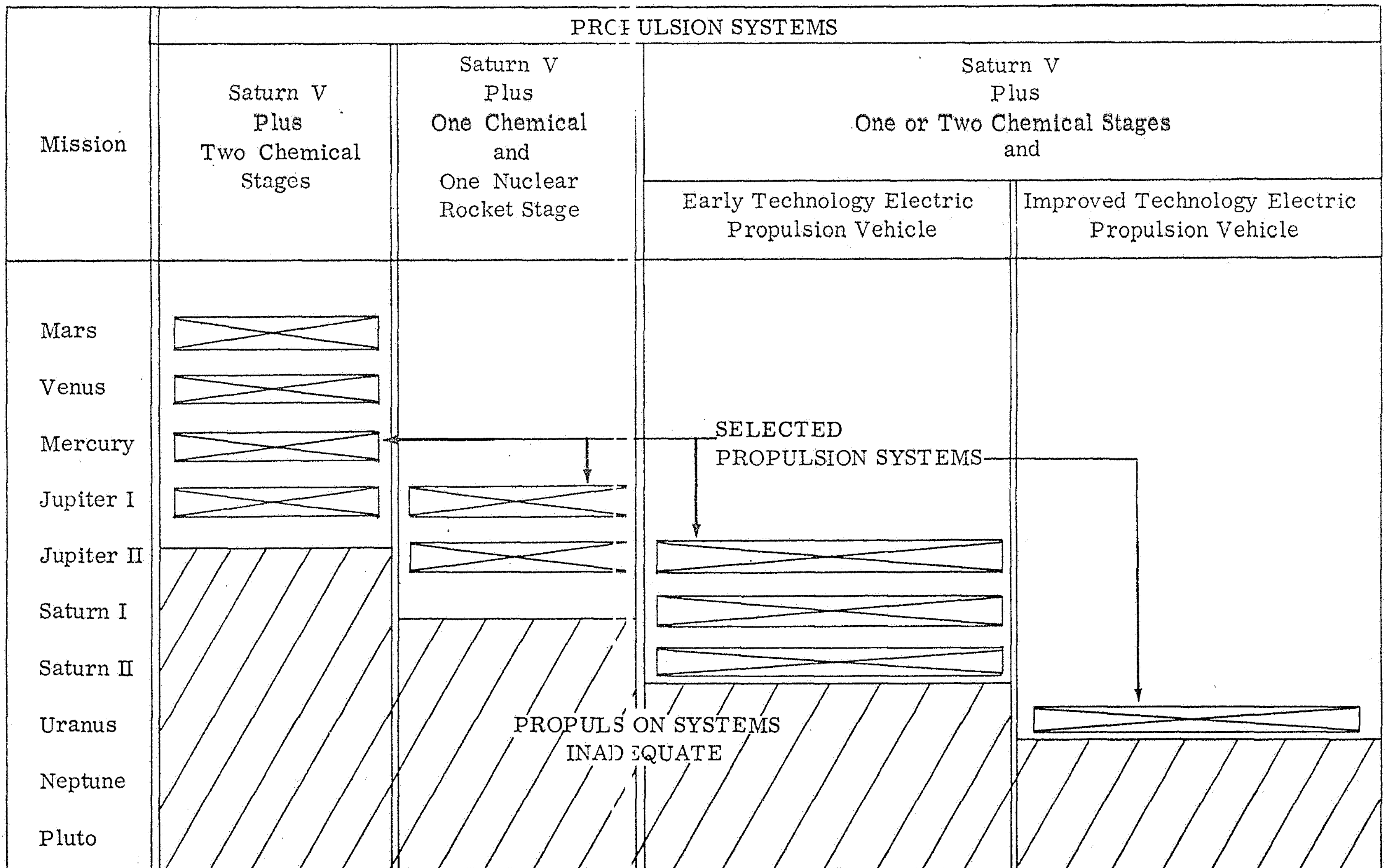


Table 3.3-5 Comparison of Propulsion Systems for Navigator "Orbiter" Missions



- The electric propulsion system is clearly superior for 5 Flyby missions (Jupiter, Saturn, Uranus, Neptune, and Pluto) and for 3 Orbiter missions (Saturn I, Saturn II, and Uranus) with possible application for the Jupiter II Orbiter Mission. All but 2 of the electric propulsion missions can be accomplished by the "Early" technology powerplant.

- An alternate propulsion profile will be required for the more distant outer planet Orbiter Missions. A S-V booster with Nuclear Rocket and electric propulsion vehicle upper stages may deliver the necessary payload (10,000 pounds) in a reasonable trip time (30,000 hours).

4. PERFORMANCE IMPROVEMENT POTENTIAL FOR ELECTRIC PROPULSION VEHICLES

The Navigator mission performance capabilities presented in the preceding section are the result of a superposition of the powerplant technology characteristics obtained from Volume 2 upon the mission performance results of Volume 1. The Volume 1 data is obtained from a double optimization process for the identification of the optimum combinations of power rating, specific impulse, and initial stage velocity to achieve maximum payload at constant trip time and powerplant specific weight. These data are based upon the use of an initial chemical rocket propulsion phase to achieve stage velocities up to 40,000 fps and upon the use of a single continuous electrical propulsion period - before the heliocentric coast for the fly-by missions and after the coast for the orbiters.

The following sections will indicate the potential for obtaining improved mission performance as a result of a selection of alternate analytical techniques or mission ground rules. These improvements are of sufficient magnitude to warrant more detailed investigations in each of these areas.

4.1 OPTIMIZATION PROCESS

The payload-power variation for those missions which are compatible with the improved technology powerplants are illustrated in Figure 4.1-1. Included are the resulting characteristics of the Asteroid, Jupiter, Saturn, and Uranus fly-by missions and the Mercury, Jupiter I and II, and Saturn I orbiter missions. The remaining missions are omitted because of excessive power requirements (Venus and Mars orbiters) or excessive powerplant weights (Uranus, Neptune, and Pluto orbiters). It is apparent that each of the missions shown can be performed with an improved technology powerplant of the order of 260 to 300 KW with the exception of the Saturn I and Mercury orbiters. The Saturn I orbiter mission was, therefore, selected to illustrate the effects of operation at non-optimum power levels and the consequences of revising the basic optimization process.

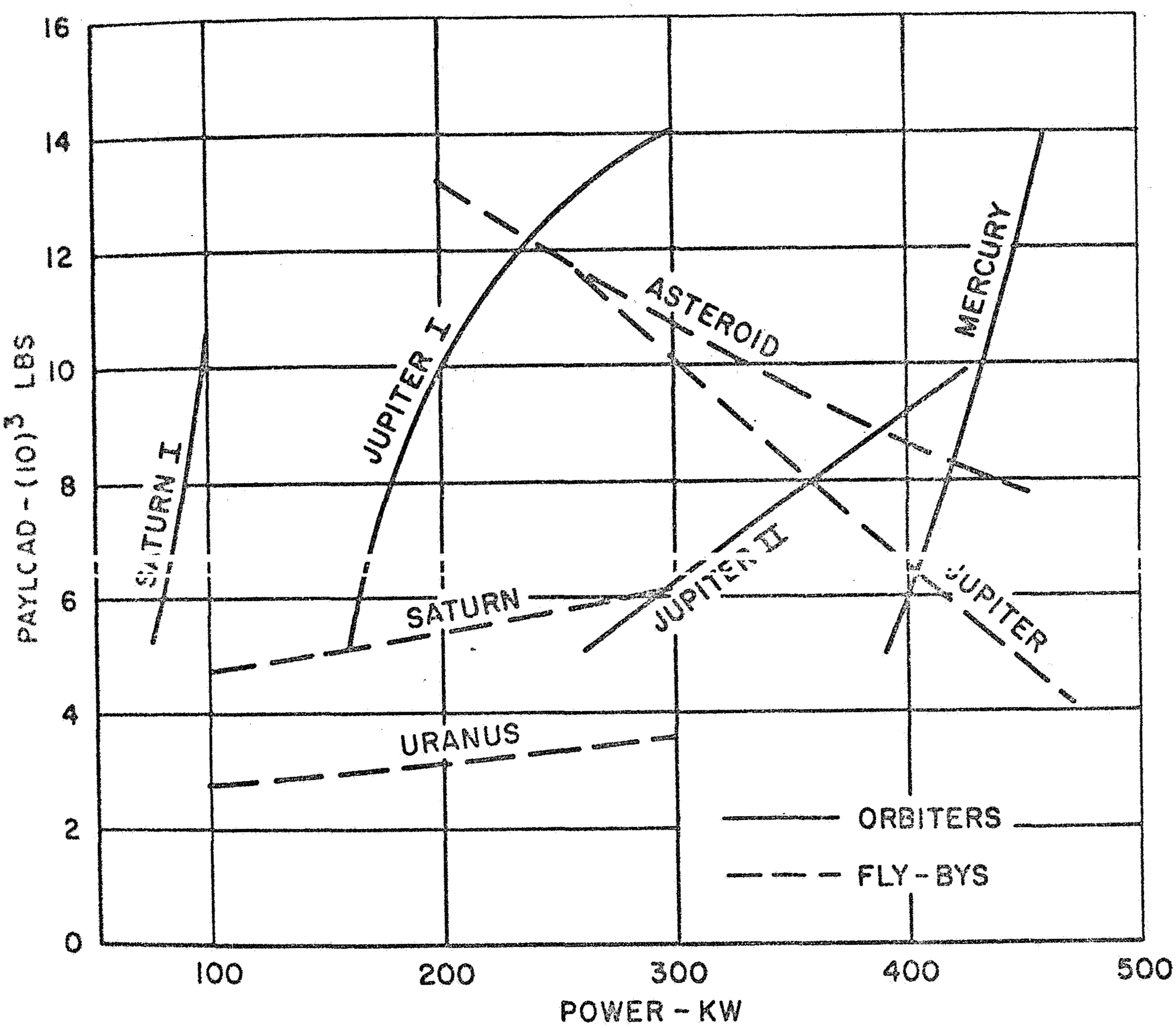


Figure 4.1-1. Navigator Payloads - Power Summary with Improved Technology Powerplants

Figure 4.1-2 illustrates the resulting Saturn I orbiter performance for an initial stage velocity of 25,000 fps with the improved level of powerplant technology. The trip time variation with power level is shown for a series of constant payload lines. Optimum operation occurs at the minimum trip time point of each payload line and represents the results of an optimization at a constant technology level. The comparable line of optimum operation obtained from the constant specific weight optimization is included as a reference. Note that the constant technology line involves a substantial increase in optimum power requirements in conjunction with some reduction in trip time requirements at constant payload. It is apparent, however, that considerable latitude is available in power level selection with only minor trip time penalty at most payload levels. Additional freedom of choice is available through variation of the initial stage velocity which has not been included in this investigation. It can be concluded, therefore, that the constant technology optimization process should be utilized in subsequent investigations.

4.2 MISSION PROFILE

The mission profiles examined in this study involve an initial high thrust acceleration with chemical rocket propulsion and a single continuous electrical propulsion period. In the case of the fly-by missions, the electrical propulsion period occurs immediately after chemical propulsion and is followed by a heliocentric coast period which lasts until the planetary fly-by occurs. The orbiter missions, on the other hand, do not initiate electrical propulsion until after the heliocentric coast period is completed. This approach in which no electrical propulsion is required before the coast period eliminates the need for either a shutdown-restart capability or for extended operation at idle power for the orbiter missions. Instead, a remote powerplant start-up after an extended soaking period is required.

An alternative is the use of an optimum coast mission profile in which the electrical propulsion operation is divided into two discrete periods, one preceding and one following the heliocentric coast period. This approach provides comparable or shorter orbiter trip times with reduced initial chemical stage velocities which, in turn, result in larger nuclear-electric spacecraft initial weights. The large initial weights permit the use of higher power

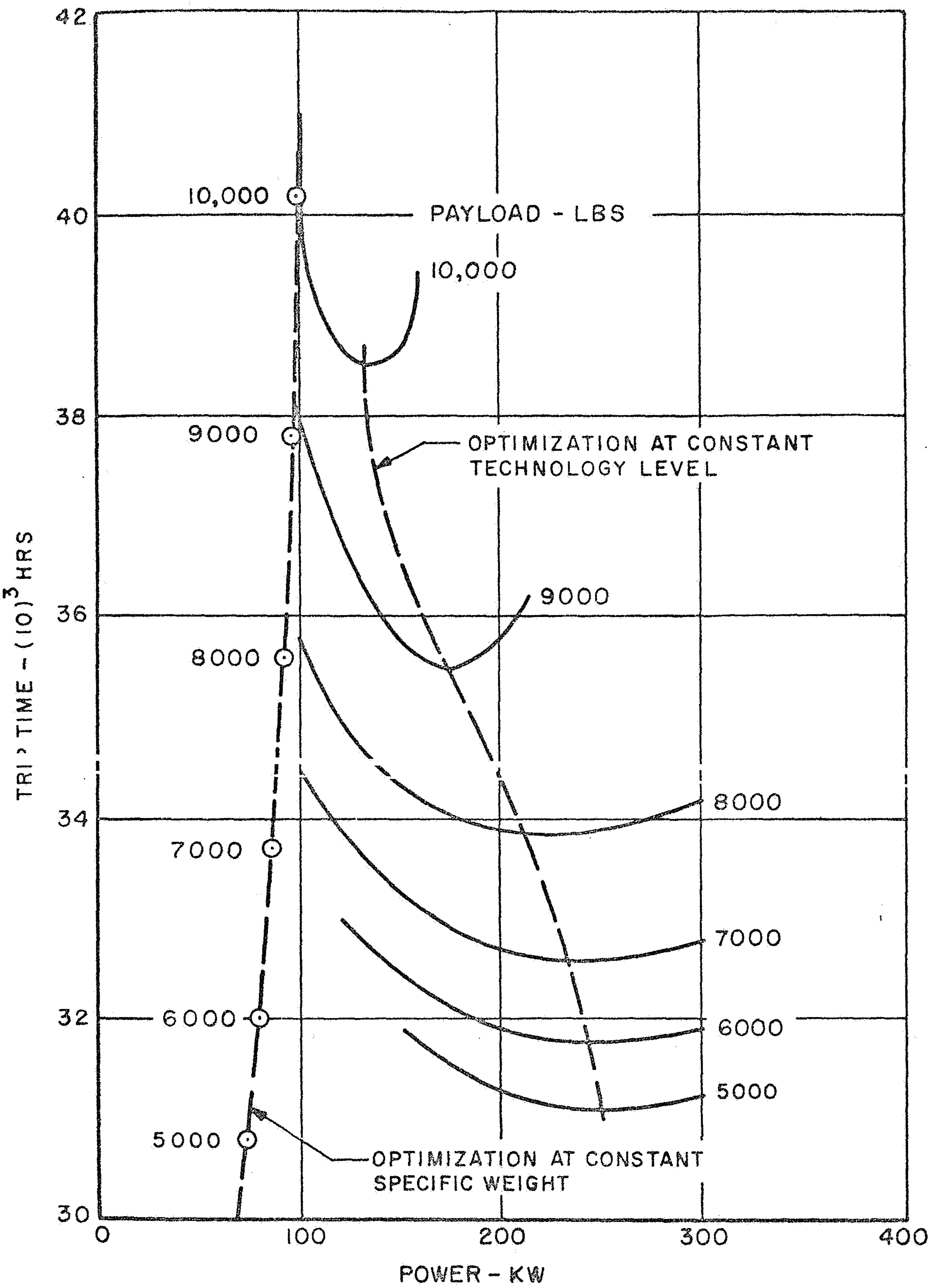


Figure 4.1-2. Saturn I Orbiter Performance with Improved Technology Powerplants and Initial Stage Velocity of 25,000 fps

levels of lower specific weights. Figure 4.2-1 illustrates typical performance capabilities for the Uranus, Neptune, and Pluto orbiter missions for a 300 KW improved technology powerplant and a 21,000 fps initial stage velocity. This is the lowest stage velocity that can be used with the Saturn V booster and still permit earth orbital test flights with the Saturn 1B. Payload is constant at 6000 lbs. The data illustrates the trade-off between propulsion and trip time. The dotted line represents comparable performance with a single propulsion period at a powerplant specific weight of 30 lbs/KW which corresponds to the improved technology specific weight at 300 KW. However, the power levels associated with the single propulsion period operation are substantially below 300 KW and, therefore, represent a more advanced level of powerplant technology. The optimum coast approach permits trip time reductions of the order of 30 to 40% at constant payload and propulsion time requirements to the level of 12,000 to 14,000 hours with no sacrifice in either payload or trip time.

The data represent an arbitrary selection of power level and initial stage velocity. Further improvements may be available by more detailed investigations designed to optimize the choice of these parameters for either minimum trip time, minimum propulsion time, or some combination of the two.

4.3 NUCLEAR ROCKET BOOST

The preceding performance data is based upon the use of three and four stage Saturn V booster configurations with LOX-LH used in the upper stages. A substantial increase in booster payload capabilities can be obtained in the 20,000 to 40,000 fps stage velocity regime by the use of a nuclear rocket of the Nerva type for the third stage in place of chemical propulsion.

Figure 4.3-1 summarizes the performance capabilities of the Saturn 5 - Nuclear Rocket for the Uranus, Neptune, and Pluto orbiter missions. These data were obtained from the continuous propulsion - constant powerplant specific weight optimization results of Volume 1 by scaling the power and payload characteristics at constant trip time and stage velocity. The scaling factors, in general, exceed 2:1 in the 20,000 to 40,000 fps regime.

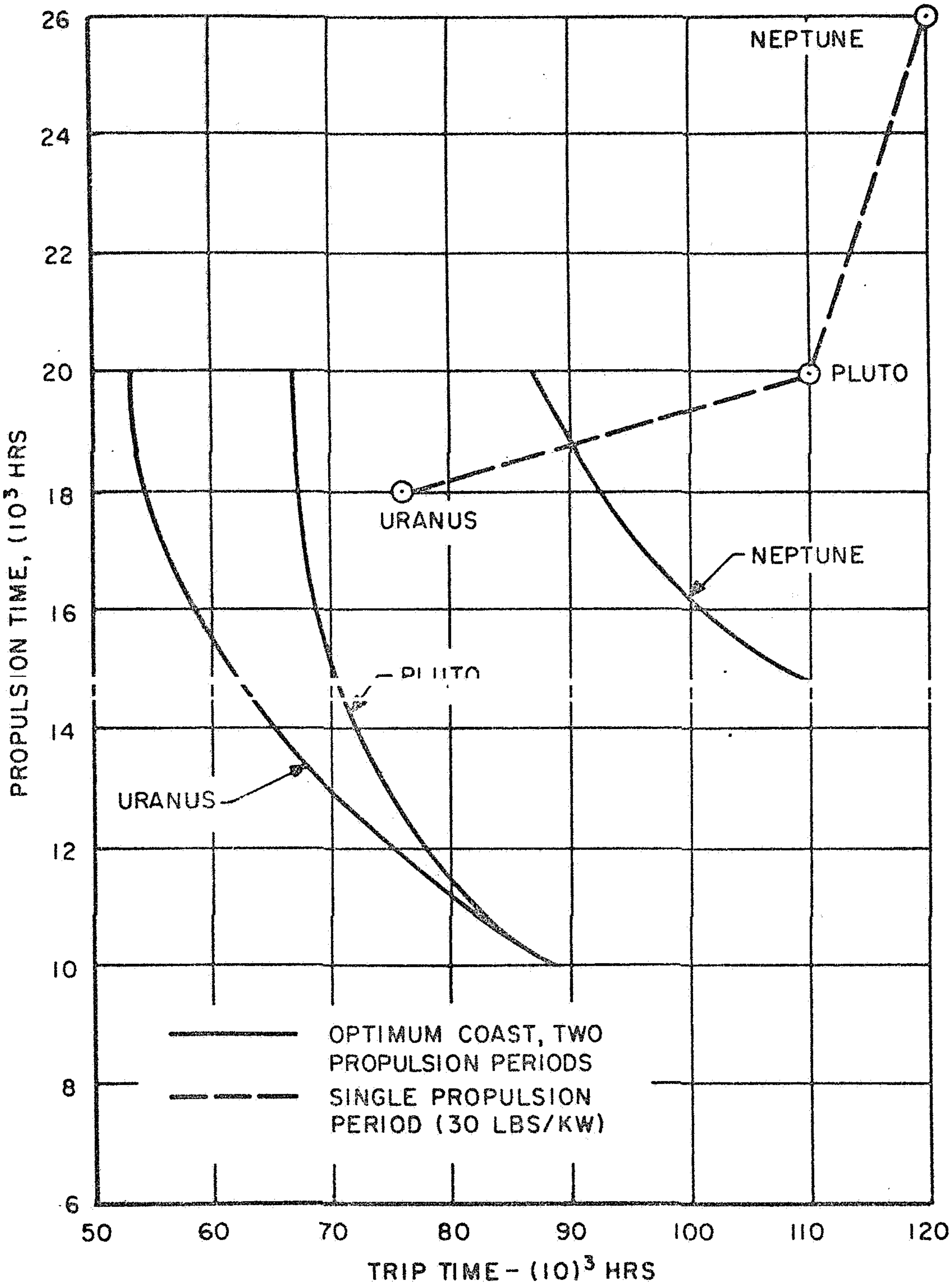


Figure 4.2-1. Outer Planet Orbiter Performance with Optimum Coast Trajectories, 300 KW Improved Technology, 6000 lbs. Payload and 21,000 Stage Velocity

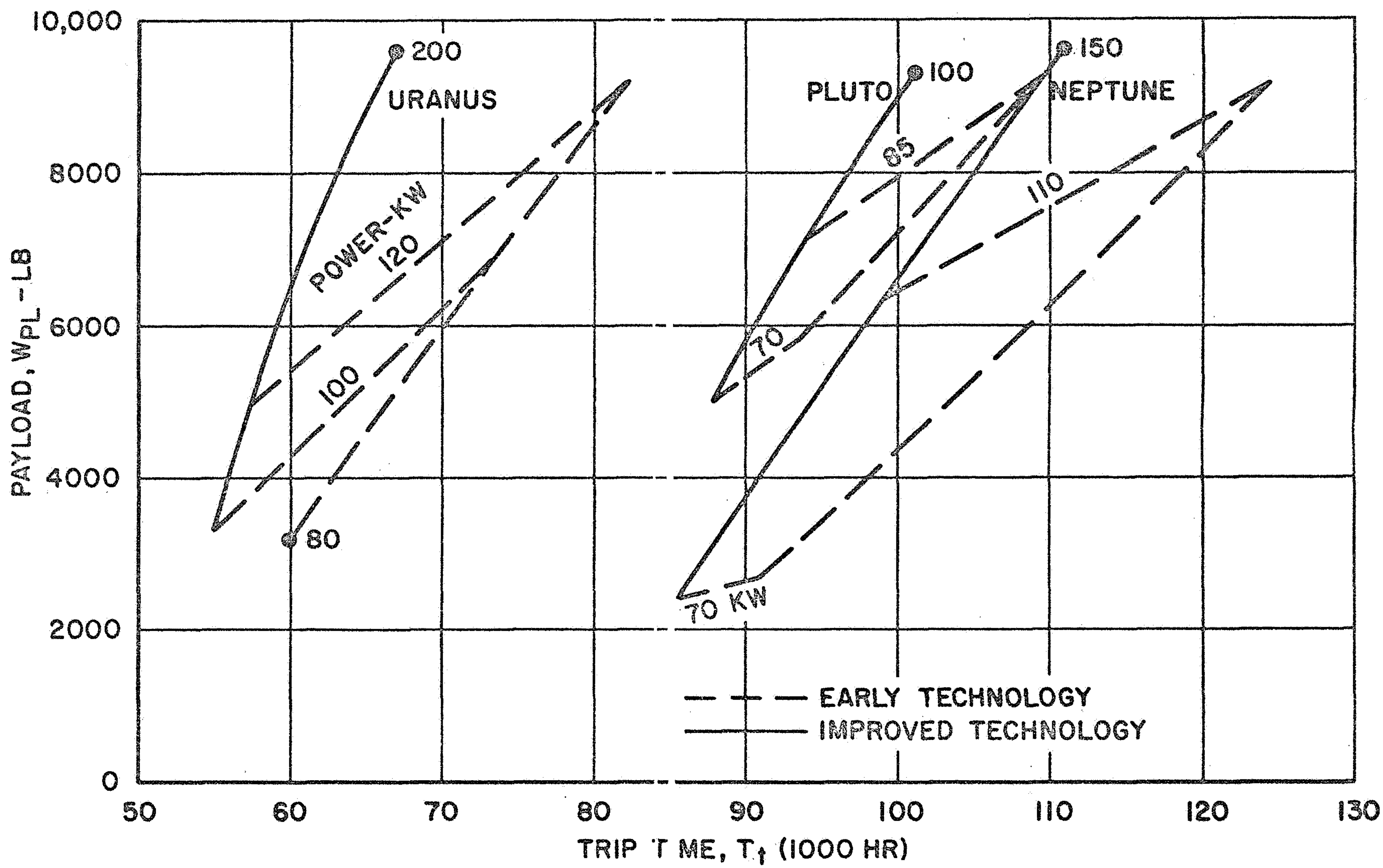


Figure 4.3-1. Major Planet Orbiter Performance with Saturn V Nuclear Rocket

resulting increase in power requirements permits matching both the early and the improved powerplant technology characteristics with the requirements for the above missions as illustrated in Figure 4.3-1. This approach results in substantial payload improvements with respect to the nominal Uranus, Neptune, and Pluto orbiter performance given in Volume 1.

Additional performance gains should be available for the above missions by combining the nuclear rocket approach with the optimum coast mission profile of Section 4.2 and with the constant technology optimization process of Section 4.1.

4.4 HIGH THRUST ORIENTATION

The orientation of the initial high thrust acceleration with respect to the sun will have a significant effect on the low acceleration propulsion requirements for the Navigator missions investigated. Recent analytical studies performed under Contract No. NAS 8-11423 (Study of Low Acceleration Space Transportation Systems) have succeeded in identifying the criteria for achieving the optimum orientation. The results of these studies were used to compare the optimum propulsion requirements with those used in the previous and present Navigator studies. It is apparent from these comparisons that some additional performance improvement may be available in the Navigator missions of interest.

Figure 4.4-1 summarizes the variation in the propulsion parameter:

$$J = \int a^2 \, dt \qquad (1)$$

with trip time for a series of optimum power limited (variable thrust) Earth-Mars trajectories. The top curve is the base point for low acceleration transfer with no initial high thrust acceleration from a circular Earth orbit to an assumed circular Mars orbit about the sun. These data are obtained by satisfying the transversality equation:

$$M = U \dot{a}_r + a_r \left[\frac{GM}{R^2} - \frac{H^2}{R^3} \right] = 0 \qquad (2)$$

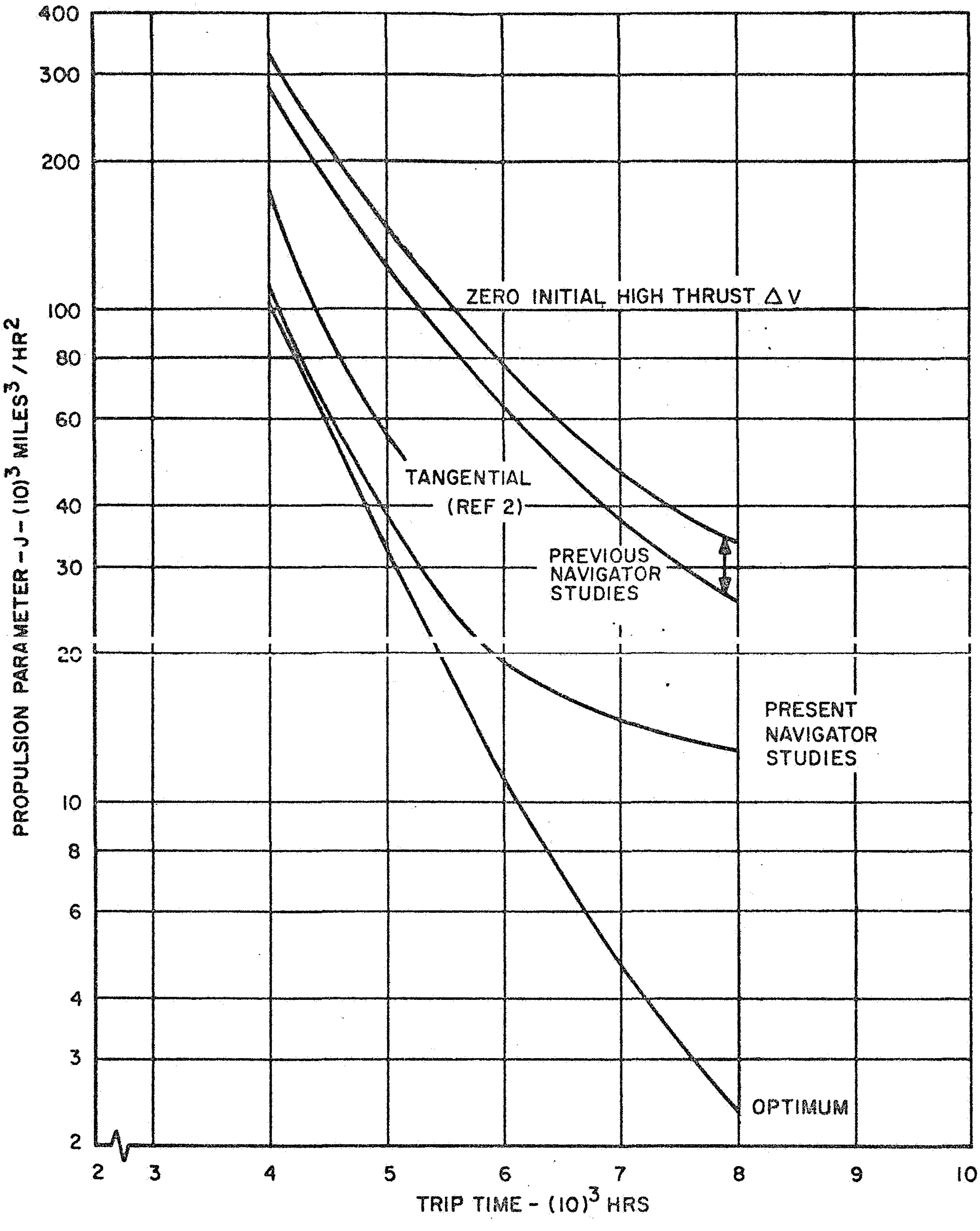


Figure 4.4-1. Effect of High Thrust-Low Thrust Matching Criteria on Mars Orbiter Propulsion Requirements. Initial High Thrust Hyperbolic Excess Velocity of 10,000 mph

at each end of the heliocentric transfer trajectory. This equation is valid for the case where the boundary orbit (initial or terminal) is either a circle or an ellipse and the boundary state variables are related by the conventional elliptical equations:

$$U = \frac{e\,GM \sin v}{H} \tag{3}$$

$$R = \frac{H^2}{GM\ [1 + e \cos v]} \tag{4}$$

The previous Navigator studies (Reference 1) were based upon the assumption that equation (2) could be used to obtain trajectories with a specified initial hyperbolic excess velocity if equations (3) and (4) were replaced at the initial point by:

$$U = V_H \cos \alpha \tag{5}$$

$$H = \sqrt{RGM} + RV_H \sin \alpha \tag{6}$$

Note that equations (5) and (6) hold only for a circular heliocentric orbit and that the angle α defines the orientation of the hyperbolic excess velocity vector with respect to the local radial direction. The resulting data are illustrated in Figure 4.4-1.

The tangential data shown in Figure 4.4-1 were obtained from Reference 2. These data are based upon the use of the following assumption:

$$\alpha = 90° \tag{7}$$

in place of equation (2). Equations (5) and (6) are maintained at the initial point and equations (2), (3), and (4) at the terminal point.

The present Navigator studies (Volume 1) employ the one-dimensional correlation technique of Reference 4 to correct the zero velocity data for the hyperbolic excess velocity. This approach produces the following correction equation:

$$J_2 = J_o - V_H \sqrt{\frac{12\, J_o}{t}} + 4\, \frac{V_H^{\,2}}{t} \tag{8}$$

The optimum data is obtained from a re-formulation of the calculus of variation problem with the initial boundary state constrained by equations (5) and (6). This approach leads to the revised transversality equation:

$$M = U\, \dot{a}_r + a_r \left[\frac{GM}{R^2} - \frac{H^2}{R^4} \right] - 1/2\; U\, a_o \sqrt{\frac{GM}{R^3}} = 0 \tag{9}$$

The optimum data is then generated by a series of trajectory calculations using equations (5), (6), and (9) at the initial boundary state and equations (2), (3), and (4) at the terminal boundary state. An empirical correlation of the difference between the optimum data and the phase 2 data indicate a variation of the form:

$$J_2 - J_o = \frac{V_H^3\, t}{R_M^{\,1.5}} \tag{10}$$

when R_M is a function of trip time and hyperbolic velocity and is of the order of 70 to 93 million miles. Although additional trajectory calculations will be required to obtain the optimum data for the Navigator missions, there would appear to be the feasibility of substantial reductions in low acceleration propulsion requirements for the high hyperbolic velocity case of interest.

5. CONCLUSIONS

This section presents a brief summary of the principal conclusions of this study. The pertinent sections from which these conclusions are derived are also given.

- The comparison of up-rated SNAP-8, Brayton Cycle and Potassium/Rankine Power Systems shows that within the technology limits specified for this study, the Potassium/Rankine Power System is nearest optimum for the Navigator Missions (Volume 2, Section 3.8).

- It appears possible to provide power system shut-down and re-start capability and to provide several kilowatts of electrical power during the coast period without significant penalty on power system weight (Volume 2, Sections 4.1, 4.2 and 4.3).

- The use of variable specific impulse can increase electric propulsion vehicle payload capability by 10 to 15% for some missions. Additional investigation is required to determine whether the performance advantage is sufficient to off-set the system complexities required to provide variable specific impulse operation (Volume 1, Section 7).

- The all chemical propulsion system will likely be adequate for the Mercury and Asteroid Belt Flyby missions and for the Mars, Venus, Mercury and Jupiter I "Orbiter" missions, (Volume 3, Section 3.3).

- The electric propulsion system will provide superior performance, will save the cost between the S-V and S-IB booster, or will be required for the Jupiter II, Saturn I, Saturn II, and Uranus Orbiter missions and for the Jupiter, Saturn, Uranus, Neptune, and Pluto Flyby missions (Volume 3, Section 3.3).

- An electric propulsion system that utilizes an early technology power system can competitively accomplish Flyby missions out to Pluto and Orbiter Missions out to Saturn. The early technology electric propulsion system can, therefore, be used to accomplish useful missions with concurrent development of improved systems for the extremely difficult planetary missions (Volume 3, Section 3.3).

- Further examination is required to evaluate the improvement in electric propulsion vehicle performance from:
  - the use of two optimum electric propulsion periods after the vehicle is launched beyond escape by the multistage S-V.
  - the selection of the optimum orientation of the high chemical thrust imparted to the vehicle.
  - the use of a nuclear rocket stage in conjunction with the electric propulsion vehicle (Volume 3, Section 4).

- Payload optimization studies at a constant powerplant technology level are more meaningful and useful than optimizations at constant powerplant specific weight (Volume 3, Section 4).

6. NOMENCLATURE

a	Low thrust acceleration, miles/hr^2.
a_o	Initial low thrust acceleration, miles/hr^2.
A_o	Coefficient of specific power equation, kw/lb thrust.
A_1	Coefficient of specific power equation, kw sec/lb thrust.
AU	Astronomical unit, solar distance divided by the mean distance of the Earth from the Sun.
	Constant thrust-optimum coast, low acceleration heliocentric trajectory optimized to minimize J with constant thrust operation. Results in intermediate coast period.
	Declination, celestial lattitude measured with respect to the ecliptic plane.
	Ecliptic plane, the plane of the Earth's orbit about the Sun.
	Fly-by trajectory, one which matches position but not velocity with target planet.
g	Sea level gravitational acceleration, 79,019 miles/hr^2.
G	Universal gravitational constant, $9.40382\ (10)^{14}$ $miles^3$/lb hr^2.
	Geocentric, central body motion with the Earth as the center of the force field.
	Heliocentric, central body motion with the Sun as the center of the force field.
	High thrust, acceleration involving thrust weight ratios greater than $(10)^{-1}$.
	Hyperbolic excess velocity, the geocentric or planetary residual velocity at infinite distance from the center of the force field.
I	Inclination angle, the angle between an orbit plane and the ecliptic plane.
I_h	Inclination angle change generated by high thrust.
I_1	Inclination angle change generated by low thrust.
I_{sp}	Thruster specific impulse, lb thrust/lb per second fuel, seconds.

J Low acceleration propulsion parameter, $miles^2/hr^3$.

L Characteristic length, measure of low acceleration propulsion requirements, miles.

L_m Minimum characteristic length, miles.

L_o Characteristic length parameter extrapolated to zero trip time, miles.

Low thrust, acceleration involving thrust weight ratios less than $(10)^{-3}$.

M_e Mass of the Earth, $1.3177\ (10)^{25}$ lb.

M_p Mass of the target planet.

M_s Mass of the Sun, $4.3894\ (10)^{30}$ lb.

N Vector normal to orbital plane.

Optimum variable specific impulse, low acceleration heliocentric trajectory optimized to minimize J at constant power. Results in large (40:1) specific impulse variation.

Orbital period, the period of revolution of an orbit.

Orbital plane, the plane defined by the instantaneous radius and velocity vectors with respect to the central body.

Orbiter trajectory, one which matches both position and velocity with the target planet and which can be converted to a low altitude planetary orbit with additional propulsion.

P Power rating, kw.

P_o Radius of orbit with respect to Earth, miles.

P_f Radius of orbit with respect to target planet, miles.

Perihelion, the point on a heliocentric orbit which is closest to the Sun.

Planetary, central body motion with the target planet as the center of the force field.

Quasi-circular, an orbit approximation in which the actual velocity is assumed to be identical with the circular orbital velocity.

R — Radius vector with respect to the Sun, miles.

R_e — Radius of the Earth's orbit with respect to the Sun, miles.

R_o — Radius with respect to the Earth, miles.

R_p — Radius of the target planet with respect to the Sun, miles.

R_t — Radius with respect to the target planet, miles.

R_∞ — The equivalent of infinite radius at which the Earth or planet no longer has any effect on the orbit.

t — Time, hr.

t_c — Coast time, hr.

t_h — Heliocentric trip time, hr.

t_m — Trip time at which characteristic length minimizes, hr.

t_p — Low acceleration propulsion time, hr.

t_{ph} — Heliocentric propulsion time, hr.

t_{pl} — Planetocentric propulsion time, hr.

t_t — Total trip time, hr.

T — Thrust, lb.

Two point boundary problem, problem involving a number of constraints at the initial and terminal ends of a trajectory which must be solved iteratively to satisfy the terminal conditions.

V — One dimensional velocity obtained by integrating acceleration in field free space or heliocentric velocity vector.

V_e — Velocity of the Earth with respect to the Sun, mph.

V_{h1} — Hyperbolic excess velocity with respect to the Earth, mph.

V_{h2} — Hyperbolic excess velocity with respect to the target planet, mph.

V_j — Thruster jet velocity, mph.

V_o Initial orbital velocity with respect to Earth, mph.

V_o Initial one dimensional velocity and equal to V_{h1}.

V_t Terminal orbit velocity with respect to planet, mph.

V_2 One dimensional velocity at coast, mph.

V_3 Terminal one dimensional velocity and equal to V_{h2}.

ΔV Low thrust characteristic velocity and equal to $g\, I_{sp}\, 1\, n\,\mu$, mph.

ΔV_{cl} Constant low thrust heliocentric characteristic velocity, mph.

ΔV_g Geocentric ΔV requirement for achieving parabolic escape from initial circular orbit at 300 miles, mph.

ΔV_h Heliocentric characteristic velocity requirement, mph.

7. REFERENCES

1. 64SD 505, Mission Analysis Topical Report, February 26, 1964, "Research on Spacecraft and Powerplant Integration Problems".

2. Private communication from C. L. Sauer of Jet Propulsion Laboratory.